# SOVIET SATELLITES AND SPACE SHIPS

(SELECTED ARTICLES)

By

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FTD-TT- 62-121/1+2

# UNEDITED ROUGH DRAFT TRANSLATION

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English Pages: 269

SOURCE: Sovetskiye Sputniki I Kosmicheskiye Korabli, 2nd Edition, Moscow 1961, Akademiya Nauk SSSR, pp. 24-82, 117-124, 144-147, 182-184, 217-242, 256-394, 427-456



PREPARED BY:

TRANSLATION SERVICES BRANCH
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# Special Features of the Motion of Satellites

By

S.G. Aleksandrov, R. Ye Federov

As a satellite moves about an elliptic orbit (Fig. 5) its height over the earth's surface h varies. In a particular case, when the height of the apogee and perigee are identical, the orbit is circular and the height of the satellite over the earth's surface is at all times constant (Fig. 6). The degree that the orbit is drawn out can be characterized by its eccentricity. The eccentricity, the semimajor axis of the orbit, the perigeal and apogeal distances are interrelated by relations (1.20) and (1.23).

It follows from these relations that the semimajor axis is equal to the average distance of the satellite from the center of the earth:

$$a = R + \frac{h_p + h_a}{2} \tag{1.32}$$

and the eccentricity of the orbit depends on the difference of heights of the apogee and perigee:

$$e = \frac{h_a - h_p}{2a} \tag{1.33}$$

Since the semimajor axis of the elliptic orbit is equal to the average distance of the satellite from the earth's center, the circling time of the satellite around the earth depends, in accordance with formula (1.31), on the average height of its flight (Table 6).

We see from the data in Table 6 that at an average flight altitude of several hundred of kilometers, the satellite circling time is about 1.5 hr, at a height of 1690 km it is 2 hr, and at a height of 35,800 km the circling time equals a sideral day (the rotation time of the earth around its axis).

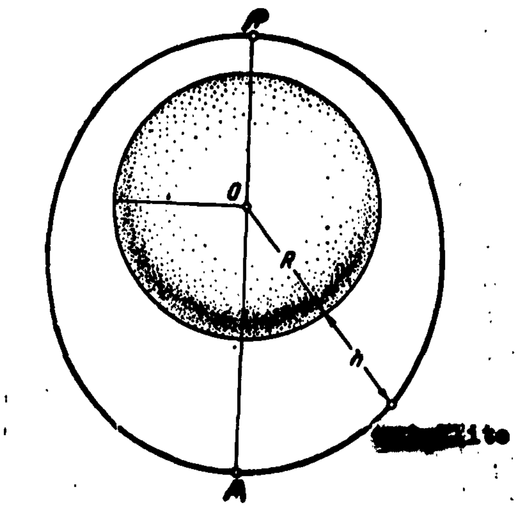


Fig. 5. Orbit of satellite
R = earth's radius; h = height of satellite over the earth's surface; P = point of perigee; A = point of apogee; O = earth's center

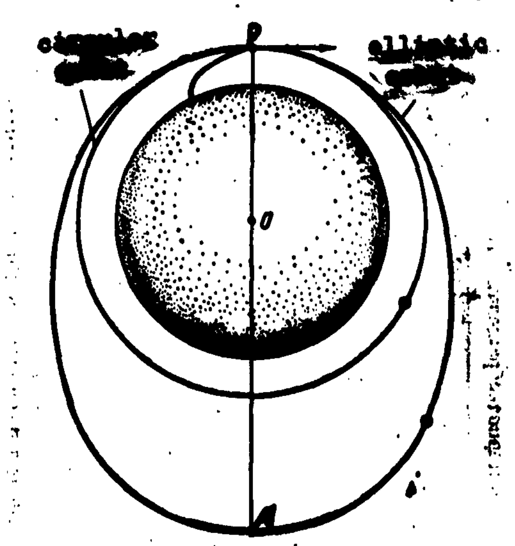


Fig. 6. Elliptic and circular orbits of satellite

A satellite launched in an eastward direction and put into a circular orbit lying in the equatorial plane at a height of 35,800 km would at all times be over one and the same point on the earth's surface.

As a satellite moves about a circular orbit its velocity is constant and equal to the circular velocity at the flight altitude of the satellite.

The values of the circular orbit for different heights are given in Table 7.

The circular velocity near the earth's surface is about 7900 m/sec.

With an increase in the height the circular velocity decreases. It is about 3070 m/sec for a height of 35,800 km (an orbit with a period of about 24 hr).

TABLE 6
Circling Time of Satellite

| Average flight altitude, km | Circling time, hr. | Average flight altitude, km | Circling time hr. |
|---|---|---|---|
| 0 | 1,41 | 1690 | 2,00 |
| 250 | 1,49 | 2000 | 2,12 |
| 500 | 1,58 | 5000 | 3,35 |
| 750 | 1,66 | 10000 | 5,78 |
| 1000 | 1,75 | 35800 | 23,935 |
| 1500 | 1,93 | | |

As a satellite move about an elliptic orbit, its velocity changes periodically, reaching a maximum value at the perigee and a minimum value at the apogee of the orbit. The velocity of the satellite at the perigee exceeds the circular velocity at the perigeal height, and the velocity at the apogee is less than the circular velocity at the apogeal height.

TABLE 7
Values of the Circular Velocity of the Satellite

| Height, km | Circular velocity, m/sec | Height, km | Circular velocity, m/sec |
|---|---|---|---|
| 0 | 7909 | 1690 | 7032 |
| 250 | 7759 | 2000 | 6901 |
| 500 | 7617 | 5000 | 5921 |
| 750 | 7482 | 10000 | 4935 |
| 1000 | 7354 | 35800 | 3072 |
| 1500 | 7116 | | |

Table 8 shows the values of the velocity of the satellite's motion at the perigee and apogee for orbits with different perigeal and apogeal heights.

TABLE 8

Values of the Satellite's Velocity at the Perigee and Apogee of Orbit

| Perigeal height, km | 250 | | | 500 | | | 1000 | | |
|---|---|---|---|---|---|---|---|---|---|
| Apogeal height, km | 500 | 1000 | 2500 | 1000 | 2000 | [illegible] | 2000 | 4000 | 10 000 |
| Velocity at perigee m/sec . . . . | 7831 | 7964 | 8304 | 7749 | 7982 | [illegible] | [illegible] | 7951 | [illegible] |
| Velocity at apogee m/sec | 7546 | 7154 | 6198 | 7224 | 6552 | [illegible] | [illegible] | 5651 | [illegible] |

The orientation of orbit in space and its position with respect to the earth's system of coordinates are usually determined by the value of the inclination of the orbit and the value of the right ascension of the ascending node. The inclination of the orbit $i$ is the angle between the orbital plane and the plane of the earth's equator. The ascending node of the orbit is the point of the orbit at which the satellite intersects the plane of the earth's equator passing from the Southern hemisphere to the Northern. Correspondingly, the opposite point on the orbit is called the descending node, and the line connecting these points is the line of nodes. The angle between the lines of the nodes and the direction to the point of the vernal equinox is called the right ascension $\Omega$.

Along with an indication of the orientation of the plane of the orbit in space, it is necessary to indicate the orientation of the orbit itself in this plane and the position of the satellite in orbit at a given instant of time. For this purpose the angular distance of the perigee from the ascending node $\omega$ (the angle between the line of nodes and the line of apsides) and the time of passage of the satellite through the orbit's ascending node $t_0$ are usually used.

Therefore the motion of a satellite about an orbit can be characterized by six elements: the inclination of the orbit $i$, the right ascension of the

ascending node $\Omega$, the semimajor axis of the orbit <u>a</u>, its eccentricity <u>e</u>, the argument of the perigee $\omega$ and the passage time of a satellite through the ascending node $t_o$ (Fig. 7).

As a satellite moves in a central gravitational field and in the absence of atmospheric drag, the first five orbital elements remain constant, and the periodic motion of the satellite in orbit can continue for an indefinitely great interval of time.

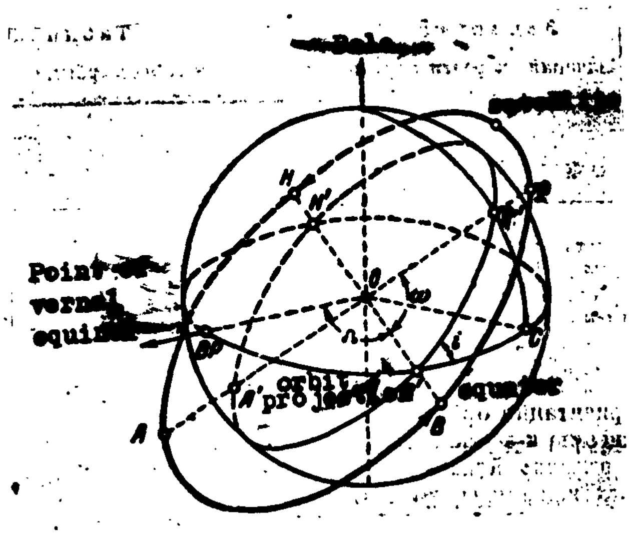


Fig. 7. Orbital elements of a satellite.

<u>i</u> is inclination; $\Omega$ is the right ascension of the ascending node; P is the orbit's perigee; P' is the projection of the perigee on the earth's surface; A is the orbit's apogee; A' is the projection of the apogee on the earth's surface; B is the ascending node of the orbit; B' is the projection of the ascending node on the earth's surface; H is the descending node of the orbit; H' is the projection of the descending node on the earth's surface; BP is the position of the point of the vernal equinox on the equator.

However, in actuality the motion of an earth satellite is influenced by a number of additional factors. These include: atmospheric drag, the difference of the earth's gravitational field and a central, fields of solar and lunar gravitation.

The effect of these factors over limited time intervals is small and thus can be considered as perturbations of the basic, Keplerian motion of

the satellite described above.

The true motion of a satellite can be represented as its motion about a Keplerian orbit whose basic elements are continuously varying, being a function of time. The current values of the orbital elements in this case are called the osculting elements.

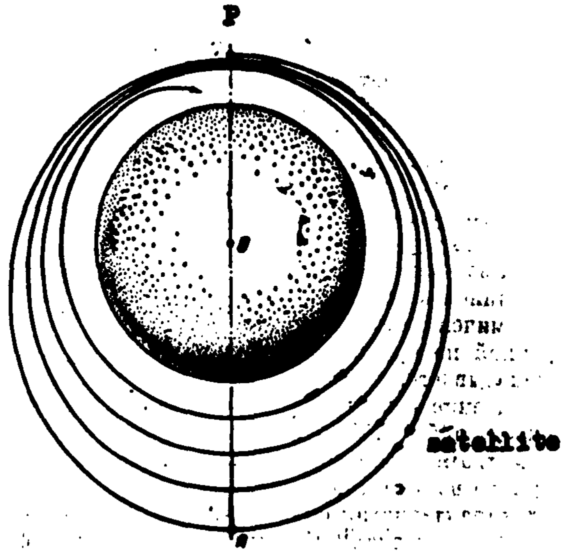


Fig. 8. Change in shape of a satellite's orbit with time due to atmospheric drag. P is the initial position of the orbit's perigee; A is the initial position of the orbit's apogee.

For satellites moving within the upper layers of the atmosphere, its resistance gradually changes the velocity of the satellite's motion and causes continuous (secular) changes in the shape of its orbit. The satellite is most strongly decelerated during the time of perigee passage. As a result of the satellite's deceleration its kinetic energy is diminished and the apogeal and perigeal distances are shortened. In addition, a decrease in the apogeal distance and height of the apogee takes place considerably more rapidly than a decrease in the perigeal distance and height of the perigee. The eccentricity is continuously reduced and the orbit itself tends to circular (Fig. 8). Deceleration of the satellite upon shortening of its orbit

progressively increases. In the end, satellite, gradually descending, enters the dense layer of the atmosphere where it is destroyed and burns owing to intense heating. After descending to an orbital height of 150 km the satellite completes no more than 1-2 orbits. The time of the satellite's motion from the instant of its going into orbit to its complete deceleration in the dense layers of the atmosphere is called its life time.

The degree of deceleration of the satellite, all other conditions being equal, depends on its aerodynamic characteristics—the drag coefficient and the ratio of weight to area of the middle (cross section), the so-called lateral load.

It is possible to establish a universal relationship between the rate of change of the height of the apogee and perigee of the orbit. This relationship is determined only by the orbital parameters and the distribution of the atmospheric density with height and does not depend on the weight and aerodynamic characteristics of the satellite. This result permits the compilation of simple tables for determining the life time of a satellite in orbit.

The life time for a satellite weighing 100 kg and having a diameter of 1 m versus the initial values of the perigeal and apogeal height of an elliptic orbit is given in Table 9. In Table 10 are the data on the life time of a similar satellite in a circular orbit.

The data in these Tables are based on the results of a theoretical investigation.*

TABLE 9
Life Time of a Satellite (in days) in Elliptic Orbits

| Height of perigee, km | Height of apogee, km | | | | |
|---|---|---|---|---|---|
| | 500 | 700 | 1000 | 1300 | 1600 |
| 200 | 9 | 18 | 37 | 58 | 82 |
| 230 | 25 | 52 | 102 | 165 | 237 |
| 260 | 53 | 116 | 236 | 370 | 535 |
| 300 | 114 | 260 | 545 | 890 | 1280 |
| 400 | 410 | 1120 | 2630 | 4450 | 6600 |

* See Uspekhi Fisicheskikh Nauk, Vol. LXIII, No. 1a, p. 33, 1957

TABLE 10

Life Time of Satellites in Circular Orbits

| Height of circular orbit | Lift time, days |
|---|---|
| 200 | 0,4 |
| 250 | 4 |
| 300 | 20 |
| 350 | 65 |
| 400 | 160 |
| 500 | 1010 |

It is apparent from the data of the Tables that for the satellite under consideration the life time at an initial perigeal height of 230 km is about 50 days. An increase in the apogeal height by 300 km (to 1000 km) leads to a doubling of the life time. About the same increase in the life time is obtained upon an increase in the height of the perigee by only 25 km (to 225 km). For circular orbits an increase in height from 300 to 400 km increases the life by about eight times, and to 500 km, another six times.

For satellites having different values of the drag coefficient and load factor on the middle, the life time, all other things being equal, is directly proportional to the value of the lateral load and inversely proportional to the drag coefficient. Thus for a satellite having a diameter of 2 m and a weight of 1000 kg, the life will increase by a factor of 2.5 in comparison with the data cited in Tables 9 and 10.

Hence is becomes evident that the dependence of the life of the satellite on the height of the orbit is very strong. At a satellite flight altitude of the order of several thousands of kilometers, its motion takes place beyond the limits of the upper layers of the atmosphere. Such a satellite can for all practical purposes be considered a permanent satellite of the earth.

The flattening of the earth at the poles and the deflection of the gravitational field from the central which is associated with this also causes perturbations of the orbit, which can be divided into peridic and secular.

The periodic perturbations of an orbit due to the non-centrality of the field are comparatively small and reduce to a deflection of the actual coordinates of the satellite from the coordinates corresponding to the motion about an ellipse, some tens of kilometers. The secular perturbations, always acting towards one and the same side, lead to substantial changes in the orbital elements with time.

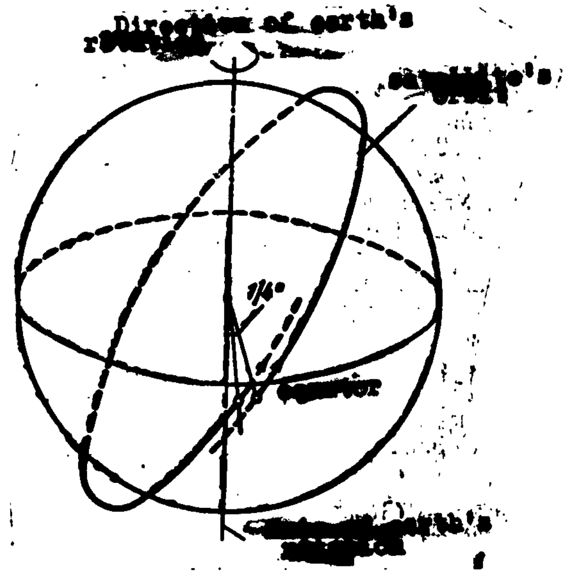

Fig. 9. Precession of an orbit in space.

The main secular perturbation of a satellite's orbit due to the non-centrality of the earth's field of gravity is the precession of the orbit, the uniform rotation of its plane in absolute space relative to the earth's axis.

The precessional velocity (the magnitude of one revolution of the satellite about its orbit) is determined by the formula:

$$\frac{d\Omega}{dN} = -\frac{2\pi R_э^2}{p^2}\left(\alpha - \frac{\Omega^2 R_э}{2g_0}\right)\cos i, \qquad (1.34)$$

where $R_э$ - the earth's equatorial radius,

$\alpha$ - is the earth's flattening,

$\Omega$ - the angular velocity of the earth's diurnal rotation,

$g_0$ - the acceleration of the earth's attraction force at the equator,



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p - the parameter of an elliptic orbit,

i - the inclination of the orbit,

N - the ordinal number of the satellite's revolution around the earth.

As follows from the formula, the precessional velocity substantially depends on the orbit's inclincation.

At an inclincation of 65° and heights corresponding to the heights of the orbit of the first Soviet satellite, the precession of the orbit is about 1/4° per revolution of the satellite (Fig. 9). At an inclination of the orbit equal to 90° (polar orbit), the precessional velocity is zero.

Another secular perturbation due to the non-centrality of the field of gravitation is the rotation of the major axis in the plane of the orbit— a change in the angular distance of the perigee from the ascending node $\omega$. There is also a shift of the perigeal region (and correspondingly, of the apogeal) from some geographic latitudes to others.

The rate of turn of the major axis of an elliptic orbit is characterized by a change in the angular distance of the perigee from the ascending node during one revolution of the satellite about the orbit:

$$\frac{d\omega}{dN}=-\frac{2\pi R_0^2}{p^2}\left(\alpha-\frac{\Omega^2 R_0}{2g_0}\right)(5\cos^2 i-1). \qquad (1.35)$$

As we see from the formula, when $i=63{,}5^\circ\ \frac{d\omega}{dN}=0.$

In accordance with formulas (1.34) and (1.35) the velocity of the precession of an orbit and the rate of drift of the perigee are inversely proportional to the square of the orbit's parameter p. Consequently, for satellites moving at considerable distance from the earth (of the order of several tens of thousands of kilometers),the precession of the orbit and the drift of the perigee due to the non-centraility of the earth's field of gravitation will be insignificant.

Therefore the flattening of the earth does not cause secular variations in the shape of the orbit. The secular perturbations of an orbit due to compression of the earth are demonstrated in the change of its orientation in absolute space. Atmospheric drag, conversely, has virtually no effect on the orbit's orientation, but does cause significant secular variations of its shape.

The effect of the solar and lunar fields of gravitation on motion of a satellite close to the earth is small, However, with an increase in the height of the orbit it increases substantially. For an orbit with an apogeal height of the order of many tens or several hundreds of thousands of kilometers, the perturbing effect of the sun and moon can cause noticeable changes in the parameters and, first of all, the perigeal height. Depending on the disposition of the orbit relative to the sun, the height of the perigee can either decrease or increase. A decrease of the perigeal height leads, in the final light, to the satellite entering the dense layers of the atmosphere, where it is destroyed. As a result of this the life of the artificial earth satellite moving about the orbit with apogeal height can prove to be quite limited.

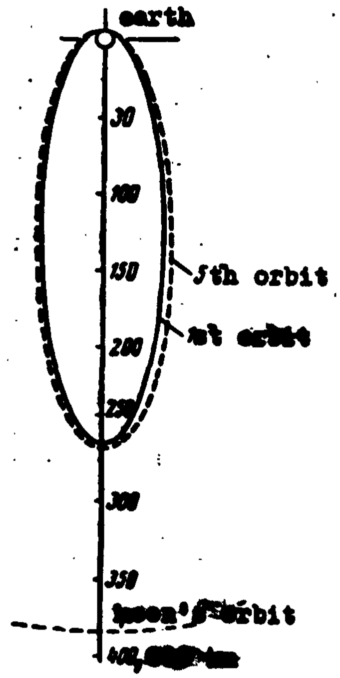


Fig. 10. Variation of satellite's orbit under the effect of the moon's gravitational field.

Thus the Soviet automatic interplanetary station after approaching the moon moved along an orbit with a perigeal height of about 40,000 km and an apogeal height of about 480,000 km. There is no atmospheric drag at such heights. Nevertheless the perturbing influence of solar attraction caused such a rapid shortening of the perigeal distance that the duration that the station moved in orbit before it entered the dense layers of the atmosphere was only a half year.

As another example, we will cite the results of the calculation of an earth satellite's motion about an orbit with an apogeal height of about 260,000 km (Fig. 10). Already during the first five orbital revolutions of the satellite its variation under the effect of perturbation by the moon's attraction is noticeable.

The motion of the satellite with respect to the earth is the result of the aggregate of its orbital motion, the diurnal rotation of the earth, and the precession of the orbits's plane.

Let us consider the motion of a satellite starting from a certain point of the orbit. During the time of one revolution of the satellite about the orbit, i.e., to the instant of its arrival at the starting geographic latitude, the earth turns to a certain angle depending on the circling time of the satellite. At the same time the plane of the orbit turns a small angle due to its precession. As a result, at the start of the next loop the satellite is over a point of the earth's surface located west of the starting point.

The projection of the motion of a satellite on the earth's surface is called the route. It is easy to show that the route of the satellite on the earth's surface passes within the limits of two parallels symmetrical with respect to the equator. These parallels correspond to the values of the northern and southern latitudes equal in value to the inclination of the orbit. Having touched one of these parallels, the route drifts toward the equator,

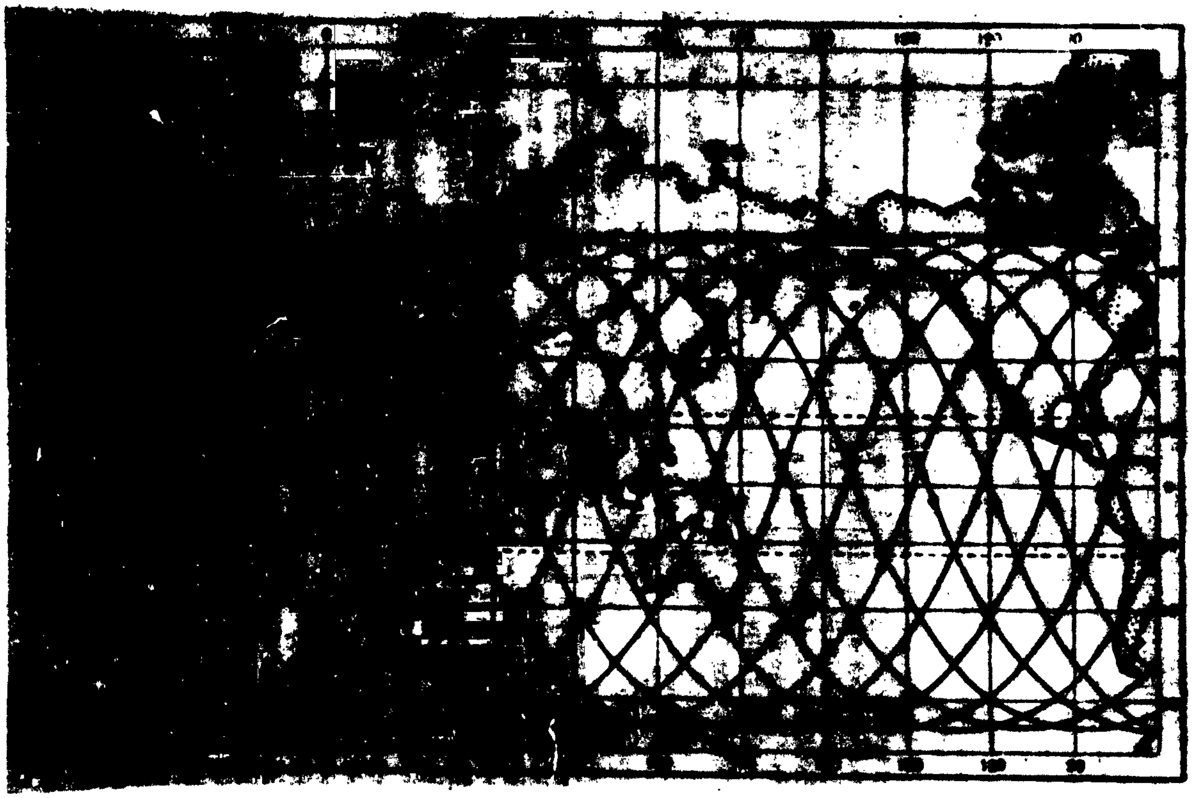

Fig. 11. Route of satellite during 24 hour period



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intersects it, then approaches the other parallel, touches it, again intersects the equator, and again approaches the first parallel.

Further, the course of the route is periodically repeated, and each of its sections corresponding to the next revolution of the satellite about the orbit is logitudinally shifted with respect to the preceding. The angle of intersection of the route with the equator somewhat differs from the angle of inclination of the orbit's plane, which is explained by the earth's rotation.

The route of the satellite during a 24-hour period with a time of revolution of about 1.6 hr is shown in Fig. 11.

The region of geographic latitudes, within the limits of which the route of the satellite passes, as was indicated above, depends on the orbit's inclination. The limiting cases with respect to the magnitude of the inclination of the orbit's plane are: a polar orbit passing through the North and South Poles and an equatorial orbit lying in the plane of the equator (Fig.12).

Artificial satellites of the moon and other planets can be created along with artificial earth satellites. The characteristics of the motion of such satellites in circular orbits (the circling time and the circular velocity) relative to the height of the orbit over the surface of the planet are shown in Table 11.

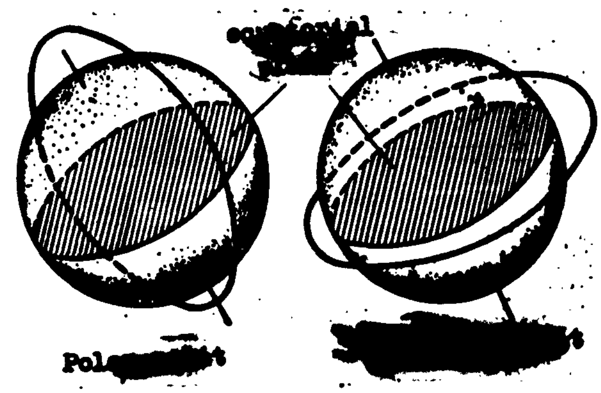

Fig. 12. Polar and equatorial orbits.



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At small heights the circling time of artificial satellites of other planets and the moon are greater than that of artificial earth satellites. This is explained by the fact that the density of other planets is less than the earth's density. The circling time of a satellite moving near the surface of a celestial body depends exclusively on the average density of this body-- it is inversely proportional to the square of density.

TABLE 11
Characteristics of the Motion of Artificial Satellites of Planets and the Moon

| Planet | Circling time hr | | | Circular velocity km/sec | | |
|---|---|---|---|---|---|---|
| | Orbital height, km | | | | | |
| | 0 | 1000 | 5000 | 0 | 1000 | 5000 |
| Mercury | 1,48 | 2,46 | 7,70 | 2,94 | 2,48 | 1,70 |
| Venus | 1,49 | 1,88 | 3,63 | 7,24 | 6,70 | 5,38 |
| Earth | 1,41 | 1,75 | 3,35 | 7,91 | 7,35 | 5,92 |
| Moon | 1,82 | 3,60 | 13,90 | 1,68 | 1,34 | 0,85 |
| Mars | 1,61 | 2,38 | 6,38 | 3,60 | 3,15 | 2,27 |
| Jupiter | 2,86 | 2,92 | 3,18 | 42,60 | 42,30 | 41,10 |
| Saturn | 3,92 | 4,01 | 4,43 | 25,65 | 25,45 | 24,60 |
| Uranus | 2,95 | 3,13 | 3,85 | 15,10 | 14,80 | 13,80 |
| Neptune | 2,63 | 2,80 | 3,46 | 16,55 | 16,20 | 15,10 |

Knowledge of the laws of motion of artificial satellites permits us to solve the problem concerning the study of the density of the upper atmospheric layers, and also the gravitational field of the earth by observing the variations in the orbit of satellites. Since there is a direct relation between the variation of a satellite's orbit and the density of the atmosphere, an analysis of the motion of artificial satellites can yield extremely valuable information on the actual values of the atmosphere's density at great heights. According to the presently available data, the density distribution of the atmosphere at great heights strongly depends on the geographic latitudes, time of day, and season of the year. Therefore, extremely important for studying the distribution of density is the joint treatment of the results of observing satellites launched at different seasons of the year and having different

inclinations of the plane of the orbit and perigeal heights.

In addition to the data on the density of the atmosphere, we can obtain from an analysis of the motion of artificial satellites, accurate values of the field strength of the earth's gravitation, and also the extent of the earth's compression and magnitude of its semimajor axis. With a sufficiently high accuracy of measuring the satellite's coordinates and with a determined lay-out of the measuring points, we can also obtain more detailed information on the earth's field of gravitation and, in particular, determine the intensity of the anomalies of the gravitational forces at different points on the earth's surface.

Besides studying the motion of the center of gravity of an artificial satellite, the study of its motion relative to the center of gravity is of considerable interest.

A non-orientated satellite having a longitudinal axis of symmetry, under the effect of perturbations taking place during its separation from a rocket carrier starts to perform a precessional motion, rotating relative to its own longitudinal axis, which in turn rotates around the precessional axis, thus forming with it a certain angle.

In the absence of atmospheric effects and other disturbing factors on the orbit, the position of the axis of precession in space remains constant relative to the stars.

However, for artificial earth satellites the position of the precessional axis in space in most cases slowly changes under the effect of aerodynamic forces and the earth's gravitational field. Furthermore, under the influence of electromagnetic forces the rotational velocity of the satellite gradually decreases. By carrying out measurements of the satellite's position at individual instants of time and by processing the results of these measurements with consideration of the equations of motion of the satellite, we can obtain

a complete picture of its motion relative to the center of gravity.

In conclusion we must note that knowledge of the current orientation of a satellite is extremely important from the point of view of setting up many scientific experiments. When analyzing the motion of a satellite in orbit it is frequently necessary to know its orientation also since with an elongated shape of the satellite the coefficient of aerodynamic drag substantially depends on its orientation relative to the velocity vector.

## Peculiarities of motion of space vehicles in the solar system
## Flights to the Moon, Mars and Venus
## Man-made planets

As shown above, when analyzing the motion of a space vehicle it was found to be possible in the first approximation to take into account the attraction of only that heavenly body in whose sphere of influence the space vehicle is located. Under these conditions the motion of a space vehicle within the sphere of influence of each heavenly body (to be examined in a coordinate system connected with this body) takes place according to one of the conic sections—ellipse, parabola or hyperbola—and is Kepler motion.

When a space vehicle reaches the boundary of the sphere of influence, the parameters of its motion must be recalculated in a new coordinate system of the heavenly body in whose sphere of influence further motion of the space vehicle is to take place.

This approximate method of studying the motion of space vehicles with respect to the individual characteristic phases, in spite of its simplicity, allows the most important laws of flight of space vehicles to be established with sufficient accuracy in many cases and also allows the fundamental characteristics of their trajectories to be determined. These cases are, in particular, flights to the Moon and other planets using chemical-fuel rockets.

Starting from these premises, let us examine the problem of flight to the Moon.

The Moon, a natural satellite of the Earth, moves about it in a near-circular orbit. The length of one orbit of the Moon about the Earth is about 27.3 days. Its distance from the Earth is on the average 384,400 km. The velocity of the Moon in orbit is about 1 km/sec. Moving at this velocity, the Moon describes in the course of one day an arc of about 13° about the celestial sphere. The plane of the orbit of the Moon is at the present time

inclined at an angle of about 18° to the equatorial plane of the Earth.

Three fundamental types of flights to the moon may be represented:

a) the landing of a space vehicle on the Moon;

b) a flight around the Moon and returning to Earth;

c) a flight close to the Moon with the subsequent ejection of space apparatus beyond the sphere of influence of the Earth, converting it into a satellite of the Sun—an artificial planet.

Each of the types of flights is of individual interest, permitting the solution of a definite range of scientific problems

Investigation of the problem of the minimum velocity necessary to reach the Moon indicates that in order to bring a space vehicle close to the Moon in the first orbit, it is necessary to assign it a velocity greater than or equal to $V_{0_{min}}$, which is a function of the altitude at the end of the ejection phase. At an atlitude of about 200 km, $V_{0min} = V_{par}$—95 m/sec (here $V_{par}$ is the parabolic velocity at the given altitude). The orbit of a space vehicle at $V_0 = V_{0min}$ is an ellipse with apoge equal to the distance from the Earth to the Moon's orbit.

A flight to the Moon in this way can be accomplished with various initial velocities; the lower the initial velocity the higher the parabolic velocity. Consequently, the trajectory of the space vehicle before entry into the Moon's sphere of influence may be elliptical, parabolic or hyperbolic.

After the space vehicle reaches the boundary of the sphere of influence of the Moon, its parameters of motion must be, in accordance with the approximate method, recalculated in the coordinate system connected with the center of the Moon—in the so-called selenocentric system of coordinates. For this the velocity vector V* of the space vehicle must be added to the vector which is the inverse of the velocity vector of the center of the Moon in the geocentric coordinate system $V_1$ (Fig. 13). The beginning of motion of the space vehicle

in the selenocentric system corresponds to the point of entry into the sphere of influence of the Moon B.

It can be shown that the trajectory of a space vehicle within the sphere of influence of the Moon in the selenocentric coordinate system is always hyperbolic. This is explained by the fact that at any flight trajectory from the Earth to the boundary of the Monn's sphere of influence, the selenocentric velocity of the space vehicle at the boundary of the sphere of influence will exceed the parabolic velocity relative to the Moon (0.383 km/sec) by at least a factor of two. Owing to this, any approach trajectory, passing by the Moon, must go beyond its sphere of influence[1].

From this comes the statement on the impossibility of the capture of a space vehicle by the Moon's gravitational field. In order to transform a space vehicle into an artificial satellite of the Moon, it must be given an additional velocity such that its velocity in the selenocentric system will be lower than the parabolic velocity. For this the space vehicle must be equipped with rocket motors to be switched on when entering the Moon's sphere of influence.

Flight to the Moon is most favorable when the plane of the trajectory of the space vehicle coincides with the plane of the lunar orbit. This can be realised by launching the rockets from the equatorial regions. In other cases, when the rocket is launched from the middle or polar latitudes, and, in particular, when launched from the USSR, the plane of its trajectory cannot coincide with the plane of the Moon's orbit. Flights to the Moon under these conditions are a more difficult problem and necessitate higher demands of power and accuracy of the control system of the space vehicle.

1 When using the approximate method the parameters of motion of the space vehicle at the point of departure from the Moon's sphere of influence must be again recalculated, in the geocentric coordinate system.

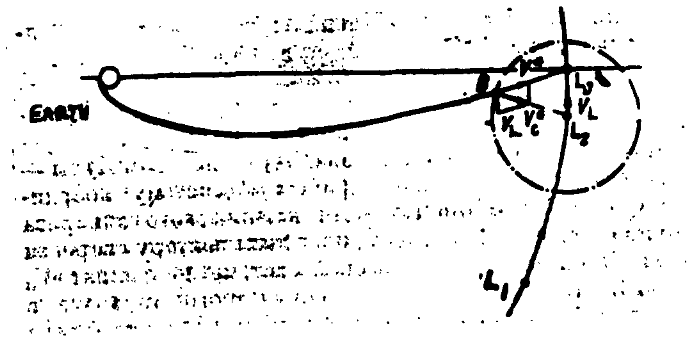


Fig. 13. Diagram of flight trajectory to the Moon lying in the Moon's orbital plane.

$L_1$—position of Moon at the time of launching; $L_2$—position of Moon when the space vehicle reaches the boundary of the Moon's sphere of influence; B—point of entry of trajectory in the Moon's sphere of influence; $\underline{V}_1$—velocity of Moon in orbit; V*—velocity of space vehicle at point B in geocentric coordinates; $L_3$—position of Moon at moment of approach from space vehicle.

Let us examine this problem in more detail. Let the rocket be launched from the Northern hemisphere of the Earth and the point A, corresponding to the end of the ejection phase, be located at latitude $\psi$ g, and the Moon at point L at the moment of impact (Fig. 14).

The orbit of the space vehicle, passing through points A and L, lies in the plane AOL, where O is the center of the Earth. The angle between the bearings OA and OL is called the range angle $\Phi$. Its magnitude is a function of the position of point A, which moves parallel to the rotation of the Earth, and also of the poisition of the Moon in its orbit at the moment of impact.

The largest value of the angle $\Phi$ is when the Moon's declination is close to −18° at the moment of impact (the Moon is below the equatorial plane), and the rocket is launched at the moment when the Moon is close to the lower

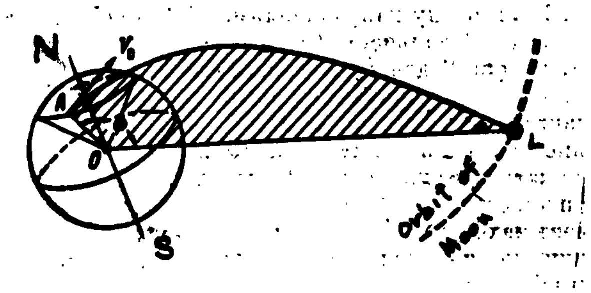


Fig. 14. Diagram of the flight trajectory to the Moon of a rocket launched from the Northern hemisphere of the Earth.

A--launching point; $V_0$--velocity at the end of the ejection phase; L--position of the Moon at the moment of impact; $\Phi$--range angle.

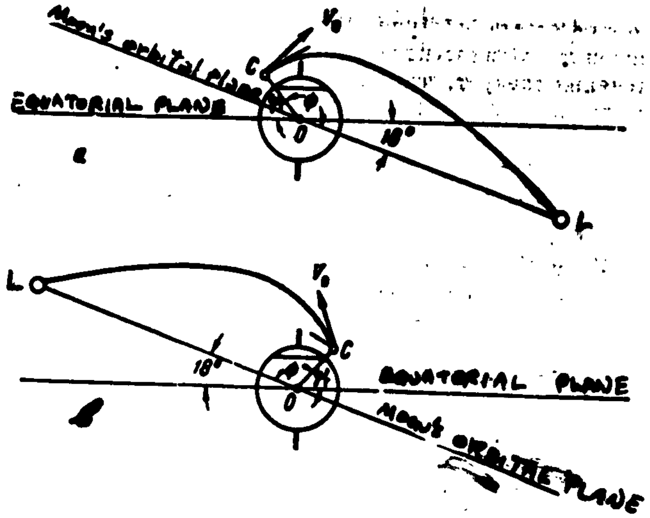


Fig. 15. Trajectory of flight to the Moon.

a--at minimum declination of the Moon at moment of impact;
b--at maximum declination of the Moon at moment of impact.

(the orbital plane of the space vehicle coincides with the Earth's axis)

culmination point (relative to the launching point). Consequently, the lowest value of $\Phi$ is when the Moon's declination is close to +18°. This may be illustrated most graphically when the orbital plane of the space vehicle coincides with the axis of the Earth, i.e., when the inclination of the orbit equals 90°. The maximum value of $\Phi$ in this case is $\Phi + 180° + 18° - \psi g$ (Fig. 15a), and the minimum is $\Phi = 180° - 18° - \psi g$ (Fig. 15b).

Now let us examine how the magnitude of the angle $\Phi$ affects flight conditions to the Moon. From the above formulas of motion of a space vehicle in the central gravitational field, it follows that its parameters of motion at the end of the ejection phase and at the point of impact must be uniquely connected with the function

$$F\left(\frac{V_0}{V_{par}}, \frac{r_0}{r_L}, \vartheta_0, \Phi\right) = 0 \qquad (1.36)$$

where $V_0$ and $V_{par}$ are the velocity of the space vehicle and the parabolic velocity at the end of the ejection phase respectively; $r_0$ and $r_L$ the distances from the center of the Earth to the end of the ejection phase and to the Moon's orbit; and $\vartheta_0$ is the angle of inclination of the velocity vector to the horizon at the end of the ejection phase.

Since the ratio $\frac{r_0}{r_L}$ is small and for all practical purposes constant, at any given value of $\frac{V_0}{V_{par}}$ the angle $\vartheta_0$ is a function only of $\Phi$. Fig. 16 shows the function $\vartheta_0 = f(\Phi)$ at various values of $V_0$ and $h_0 = 200$ km.

As is apparent from the graph, for all values of $V_0$ the magnitude of $\vartheta_0$ decreases substantially with an increase in $\Phi$.

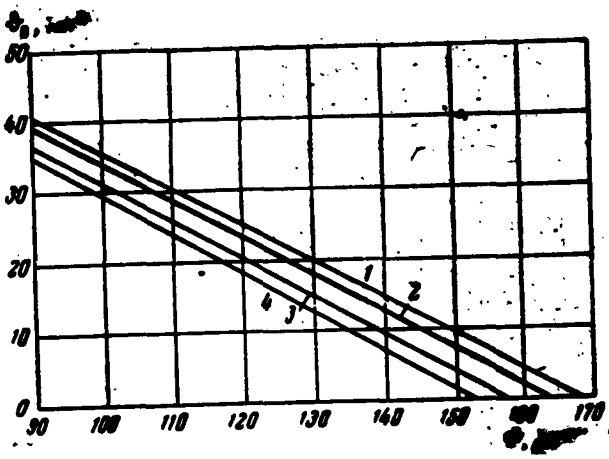


Fig. 16. The dependence of the angle of inclination of the velocity vector $\vartheta_0$ upon the range angle $\Phi$.

1—at $V_0 = V_{par}$ −50 m/sec; 2—$V_0 = V_{par}$; 3—$V_0 = V_{par}$

3—$V_0 = V_{par}$ + 100 m/sec; 4—$V_0 = V_{par}$ + 200 m/sec.

The decrease in $\vartheta_0$, i.e., transition to more sloping ejection trajectories, entails a lowering of losses in overcoming the gravitational forces $\int_0^t g \sin \vartheta_0 \, dt$ and permits increasing the pay load of the rocket.

From this it follows that when launching the space vehicle from the middle latitudes of the Northern hemisphere, from the point of view of power, it is more advantageous to launch the rocket when the Moon is near its orbital point of minimum declination. In this case it is possible to put the maximum payload into orbit. When launching a rocket at an earlier or later date, the maximum possible payload is decreased. However, when the launching date deviates from the optimum date by several days, the decrease in payload is comparatively small. In practice, in the course of each lunar month there are intervals of time of about a week in which flight conditions to the Moon are favorable. When the dates greatly deviate from the optimum, considerably lower payloads are possible.

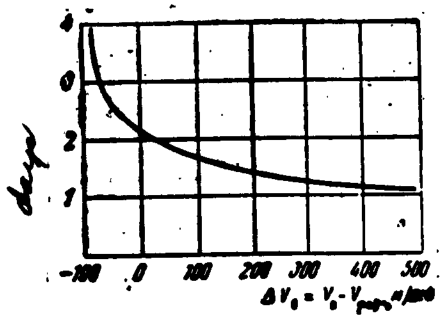


Fig. 17. Duration of flight of space vehicle to the Moon ($h_0$ = 200 km).

The duration of a flight to the moon is determined by the rocket's velocity relative to the parabolic velocity (Fig. 17). At minimum initial velocity ($V_0 = V_{min}$) the flight duration is slightly over four days, at $V_0 = V_{par}$ it is about two days, and at a velocity exceeding the parabolic velocity by 500 m/sec, it is about one day.

When making flights to the Moon, it is in most cases necessary to observe the space vehicle and to receive telemetric information while the vehicle is approaching the Moon and at the moment of impact. For this it is necessary that the Moon during this period be located in relation to observation points near the upper culmination point. At the same time, as shown above, considering power it is advantageous to launch the rocket when the Moon is located near the lower culmination point relative to the launching point. It is obvious that both of these conditions can be fulfilled only if the flight duration is a multiple of one half day, i.e., one half day, one and one half days, two and one half days, etc. The most advantageous is a flight of about one and one half days, for which the velocity of the space vehicle at the end of the ejection phase must somewhat exceed the parabolic velocity (by approximately 150 m/sec). A flight with a duration of about one half day requires considerably exceeding of the velocity at the end of the ejection phase over the parabolic velocity, while flights of two and one half or more days require more accurate ejection of the vehicle into orbit for striking the Moon.

For flights around the Moon with return to Earth, the velocity of the space vehicle at the end of the ejection phase must be less than the parabolic velocity. Depending upon the degree of nearness to the Moon, two basic types of flights may be distinguished--far and near flights around the Moon.

In a far flight around the Moon, when the minimum distance from the space vehicle to the Moon is 40,000 km or more, the effect of the Moon's gravitational field upon the space vehicle's motion is not great, and its orbit in the geocentric system will be close to elliptical. If the rocket is launched from the middle latitudes, the angle of inclination of the velocity vector at the end of the ejection phase will be substantially different from zero and, therefore, the elliptical orbit will intersect the Earth's surface. As a result, the space vehicle will at the end of the first orbit pass through the dense layers of the atmosphere and be destroyed.

Of special interest is the so-called near flight around the Moon, when the minimum distance of the space vehicle from the Moon is about 5 to 10 thousand km. In this case it is possible, using the gravitational force of the Moon, to change the nature of the orbit of a space vehicle in the period of its first approach to the Moon in such a way as to obtain a new orbit, corresponding to the new requirements. In particular, this change in orbit can be ensured so that the space vehicle will be transformed into an artificial Earth satellite, moving in an orbit with perigee equal to several tens of thousands of kilometers, and apogee reaching one half million kilometers. An example of this type of flight is the flight of the third Soviet cosmic rocket with an automatic interplanetary station.

Let us now examine the case when the trajectory of the space vehicle leaves the limits of the sphere of influence of the Earth. In these cases the velocity at the end of the ejection phase exceeds or equals the parabolic velocity, and the trajectory within the sphere of influence of the Earth is hyperbolic

or parabolic.

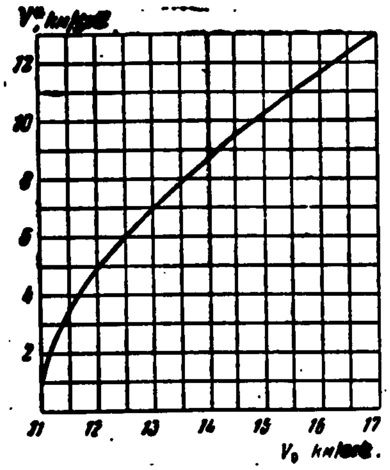

Fig. 18. The dependence of the velocity of a space vehicle at the boundary of Earth's sphere of influence upon its velocity at the end of the ejection phase ($h_0$ = 200 km).

The velocity of a space vehicle at the boundary of the Earth's sphere of influence V*, as is apparent from Formula (1.18), is uniquely determined by the velocity $V_0$ and altitude $h_0$ at the end of the ejection phase. The dependence of V* upon $V_0$ at $h_0$ = 200 km is shown in Fig. 18. When the velocity at the end of the ejection phase is equal to the parabolic velocity ($V_0$=$V_{par}$), the velocity at the boundary of the sphere of influence V* $\simeq$ 0.9 km/sec. Increasing $V_0$ by 0.5 km/sec relative to $V_{par}$ leads to an increase in V* to 3.3 km/sec, and an increase of 1 km/sec will lead to an increase in V* to 4.9 km/sec. At velocities close to the parabolic velocity an increase in $V_0$ by 1 km/sec leads to an increase in V* by 5 to 8 m/sec.

The motion of a space vehicle after leaving the Earth's sphere of influence may be calculated starting from its parameters of motion at the point of departure. The velocity of a space vehicle in heliocentric coordinates may be obtained by summation of the velocity vector $\overline{V}$* and the velocity vector of the Earth's center at the corresponding moment in time $\overline{V}_e$. This velocity vector $\overline{V}_{0c}$ also determines the nature of the motion of a space vehicle in the heliocentric system.

If $V_{0c}$ is less than the parabolic velocity relative to the Sun $V_{par_c}$, the motion of a space vehicle in the solar system will be in an elliptical orbit, and it will be transformed into a solar satellite—an artifical planet. If $V_{0c}$ is equal to the parabolic velocity $V_{par_c}$ or exceeds it, the motion of a space vehicle will be in a parabolic or hyperbolic trajectory. In these cases it will leave the solar system forever.

Let us determine the minimum velocity which must be given the space vehicle for this when leaving the Earth. The average velocity of the Earth is $V_e$ = 29.75 km/sec and, therefore, the parabolic velocity relative to the Sun, calculated taking into account the average radius of the Earth's orbit, equals $V_{par_c} = \sqrt{2}\, V_e = 42$ km/sec.

Let us assume that the velocity vector of the space vehicle at the boundary of the sphere of influence is parallel to the velocity vector of the Earth's orbital motion, which may be ensured by the appropriate choice of the launching direction of the space vehicle. Under these conditions the space vehicle will move in the heliocentric system with the parabolic velocity if its velocity in the geocentric system is $V^* = V_{par_c} - V_e = 12.25$ km/sec, for which its velocity at the end of the ejection phase, as is apparent from Fig. 18, must be about 16.5 km/sec.

The velocity necessary to take a space vehicle beyond the Sun's gravitational field is called the third cosmic velocity. It magnitude at altitude $h_0 = 0$, i.e., at the Earth's surface, is about 16.7 km/sec.

Let us examine in more detail the motion of a space vehicle in the solar system in an elliptical orbit and, in particular, flights to other planets. In the first approximation let us assume that the motions of the Earth and other planets is in a circular orbit whose radii correspond to the average radii of their actual orbits. Let us assume that the orbits of all planets and the trajectorise of the space vehicles lie in the same plane.

Let the trajectory of the space vehicle in the Earth's sphere of influence be taken such that the velocity vector at the boundary of the sphere of influence $\overline{V}^*$ is parallel to the Earth's velocity vector $\overline{V}_e$. If the directions of these coincide, the velocity of the space vehicle in heliocentric coordinates will equal their arithmetical sum and be a maximum for the given value of the velocity $V^*_0$. The orbit of the space vehicle in heliocentric coordinates in this case will envelope the Earth's orbit, tangent to it in the perihelion at a distance from the Sun of the radius of the Earth's orbit $r_\pi = r_e$. The distance from the Sun to the aphelion of the orbit $r_a$ will be a function of the velocity of the space vehicle. Values of $r_a$ as a function of the velocity at the end of the ejection phase $V_0$ at $h_0$ = 200 km are shown in Table 12.

TABLE 12

Distance from the Sun to the aphelion of the orbit at various values of the velocity at the end of the ejection phase.

| Velocity at end of ejection phase in km/sec | Excess velocity over the parabolic in km/sec | Distance from Sun to aphelion of orbit in millions of km |
|---|---|---|
| 11,015 | 0 | 168,9 |
| 11,515 | 0,5 | 247,7 |
| 12,015 | 1,0 | 314,1 |
| 13,015 | 2,0 | 480,1 |
| 14,015 | 3,0 | 760,3 |
| 15,015 | 4,0 | 1400,0 |
| 16,015 | 5,0 | 4618,0 |

From these data it is apparent that when the velocity of the space vehicle exceed the parabolic velocity by 0.5 km/sec, the aphelion of its heliocentric orbit will be located beyond the orbit of Mars, at a velocity exceeding the parabolic by 3 km/sec, it will be close to the orbit of Jupiter, and at a velocity exceeding the parabolic by 4 km/sec, it will reach the orbit of Saturn.

If the direction of the velocity vector of the space vehicle at the boundary of the sphere of influence is opposite that of the velocity vector of the

Earth, the velocity of the vehicle in the heliocentric system will equal their difference and be a minimum for the given value of $V^*_0$. In this case the orbit of the space vehicle will be within the Earth's orbit, tangent to it in the aphelion ($r_a = r_e$).

The distances from the Sun to the perihelion of an orbit of this type at the end of the ejection phase $V_0$ at $h_0 = 200$ km are given in Table 13.

As is apparent from the table, when the velocity of the space vehicle $V_0$ exceeds the parabolic velocity by 2 km/sec, the perihelion of its orbit will be close to the orbit of Mercury. To bring the orbit of the space vehicle close to the Sun it is necessary to assign it a very high velocity at the end of the ejection phase. To attain a distance of 30 million km from the Sun, the sphace vehicle must have a velocity of 16.7 km/sec, exceeding the third cosmic velocity.

TABLE 13

Distance from the Sun to the perihelion of the orbit at various velocities at the end of the ejection phase.

| Velocity at the end of ejection phase in km/sec | Excess velocity over parabolic in km/sec | Distance from Sun to perihelion of orbit in millions of km |
|---|---|---|
| 11,015 | 0 | 132,8 |
| 11,515 | 0,5 | 95,6 |
| 12,015 | 1,0 | 80,3 |
| 13,015 | 2,0 | 61,8 |
| 14,015 | 3,0 | 49,8 |
| 15,015 | 4,0 | 40,9 |
| 16,015 | 5,0 | 33,9 |

Thus, approaching the Sun at close distances presents a greater power problem than going beyond the Sun's gravitational field.

These orbits of space vehicles may be used for flights to other planets of the solar system. The minimum necessary velocity for a flight to a given planet will correspond to a semielliptical orbit, which in its aphelion and perihelion is tangent to the orbit of the Earth and the planet named. The range angle of the orbital phase of the flight in the heliocentric system will

be $\Phi_1 = 180°$ (Figs. 19 and 20).

TABLE 14

Values of the minimum possible velocities for flights to the planets ($h_0 = 200$ km)

| Planet | Velocity at end of ejection phase in km/sec | Flight duration in years |
|---|---|---|
| Mercury | 13.31 | 0.29 |
| Venus | 11.25 | 0.40 |
| Mars | 11.35 | 0.71 |
| Jupiter | 14.05 | 2.72 |
| Saturn | 15.05 | 6.04 |
| Uranus | 15.73 | 16.0 |
| Neptune | 16.00 | 30.6 |

The values of velocities necessary for flights to other planets in semi-elliptic orbits and the duration of these flights, calculated for the average radii of the orbits of the planets, under the assumption that all orbits lie in one plane, are shown in Table 14.

TABLE 15

Orbits of a flights to Mars

| Initial velocity, for heliocentric orbit, km/sec | Range angle, degrees | Flight duration, months |
|---|---|---|
| 32,71 | 180 | 8,63 |
| 33,71 | 124 | 5,25 |
| 34,71 | 108 | 4,32 |
| 35,71 | 97 | 3,77 |
| 36,71 | 90 | 3,40 |
| 37,71 | 85 | 3,10 |

At an increase in the velocity $V_0$ in comparison with the minimum required, the intersection of the orbit of the space vehicle with the orbit of a planet will take place at a lower value of the range angle $\Phi_2 < \Phi_1$. Simultaneously, the duration of the flight is increased. The basic data on these orbits for flights to Mars and Venus are given in Tables 15 and 16.

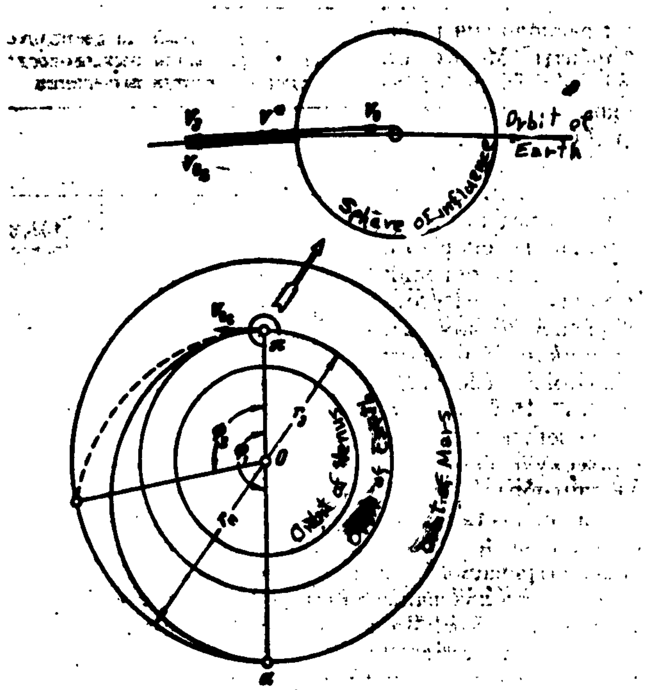


Fig. 19. Trajectory of flight to Mars.

O--Sun; $r_e$--average radius of Earth orbit; $r_M$--average radius of orbit of Mars; $V_0$--velocity of space vehicle at end of ejection phase; $V_e$--velocity of Earth in orbit; $V^*$--velocity of space vehicle at boundary of the Earth's sphere of influence in geocentric system; $V_{0_c}$--velocity of vehicle at boundary of Earth's sphere of influence in heliocentric system (initial velocity for heliocentric orbit) $\pi$ --perihelion of heliocentric orbit; $\alpha$--aphelion of heliocnetric orbit; $\Phi_1$, $\Phi_2$ --range angle for heliocentric orbits.

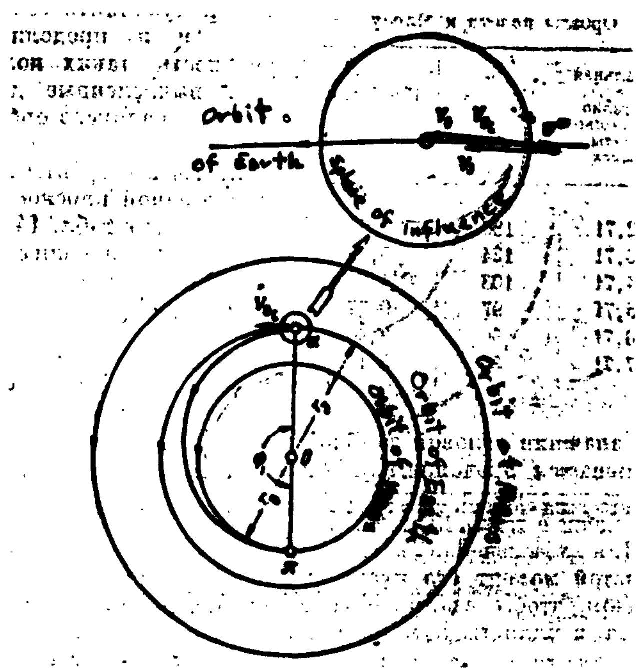


Fig. 20. Trajectories for flight to Venus.

O--Sun; $r_e$--average radius of Earth's orbit; $r_V$--average radius of orbit of Venus; $V_0$--velocity of space vehicle at end of ejection phase; $V_e$--velocity of spade vehicle at boundary of Earth's sphere of influence in geocentric system; $V_{0_c}$--velocity of vehicle at boundary of Earth's sphere of influence (initial velocity for heliocentric orbit) in heliocentric system; $\pi$--perihelion of heliocentric orbit; $\alpha$--aphelion of heliocentric orbit; $\Phi_1$--range angle.

In order that the space vehicle make contact with the planet, the time of its launching must be chosen such that the mutual position of the Earth at the moment of launching and the planet at the moment of impact be fully determined.

TABLE 16

Orbits of a flight to Venus

| Initial velocity for heliocentric orbit, km/sec | Range angle, degrees | Flight duration, months |
|---|---|---|
| 27,28 | 180 | 4,87 |
| 26,28 | 110 | 3,33 |
| 25,28 | 80 | 2,83 |
| 24,28 | 76 | 2,52 |
| 23,28 | 66 | 2,33 |
| 22,28 | 59 | 2,16 |

The favorable mutual positions of the planets are periodically repeated. For flights to Mars, their repetition period is 2.14 years, and for flights to Venus, 1.57 years.

It should be noted that this data on the required velocities for flights to the planets, owing to assumptions made in their calculation, are approximate and describe the lower limit of required velocities. They are valid when impact takes place near the node of the planet's orbit, i.e., when the space vehicle's motion is in the plane of the eclipse.

In other cases, when the space ship makes contact with a planet in a period when it is not in the plane of the eclipse (due to the fact that its orbital plane makes some angle with the plane of the eclipse), the velocity necessary for a flight to the planet is found to be considerably greater.

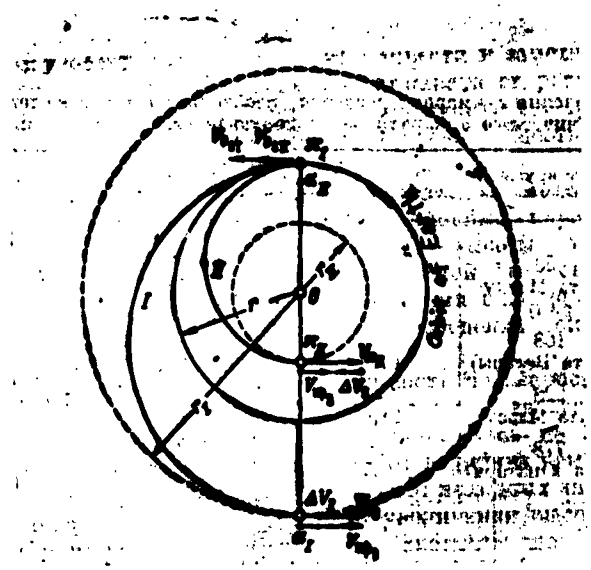

Fig. 21

Trajectories of transition of space vehicle to circular orbit around the Sun.

O—Sun; $V_{ooI}$, $V_{ooII}$—velocity of space vehicle at boundary of the Earth's sphere of influence in heliocentric system; $V_{aI}$, $V_{\pi II}$—velocity of space vehicle at point of transition to circular orbit about the Sun; $\Delta V_I$, $\Delta V_{II}$—additional velocity necessary to put vehicle in circular orbit; $V_{kpI}$, $V_{kpII}$—circular velocity; $r_I$, $r_{II}$—radius of circular orbit; r—average radius of Earth's orbit. Subscript I pertains to space-vehicle orbit of greater radius than that of Earth, II denotes orbit smaller in radius than Earth's.

TABLE 17

Values of total velocity necessary for ejection of space vehicle into circular orbit about the Sun.

| Radius of circular orbit, million km | Total velocity, km/sec | Radius of circular orbit, million km | Total velocity, km/sec |
|---|---|---|---|
| 58 (Orbit of Mercury) | 23.0 | 1426 (Orbit of Saturn) | 20.5 |
| 108 (Orbit of Venus) | 14.0 | 2869 (Orbit of Uranus) | 20.4 |
| 228 (Orbit of Mars) | 14.0 | 4495 (Orbit of Neptune) | 20.1 |
| 788 (Orbit of Jupiter) | 19.7 | | |

In conclusion, let us examine the problem of creating artificial satellites of the Sun (artificial planets) which move in circular orbits. For this the space vehicle must be first of all be placed in a semielliptical transitory orbit, tangent to the aphelion or perihelion of the given circular orbit. When the space vehicle reaches the aphelion (or perihelion) of the transitory orbit, it must be given an additional velocity $\Delta V$ for its transition into circular orbit (Fig. 21).

Values of the total velocity $V_0 + \Delta V$ necessary to put the space vehicle into a circular orbit, transforming it into an artificial planet, are shown in Table 17.

## Requirements on the Accuracy of the Motion Parameters at the End of the Ejection Phase

For a space apparatus to move along a given orbit it must have definite motion parameters. Even slight errors in the magnitude and direction of the velocity at the end of the ejection phase can result in considerable deviations of the orbit of the vehicle from the calculated orbit.

Let us demonstrate the extent to which these errors influence the orbital characteristics and the lifetime of artificial earth satellites.

As illustration, Table 18 gives data on the change of the basic parameters of the orbit of an artificial satellite, the heights of the perigee and the apogee, when there are errors in the velocity and the angle of inclination of the tangent to the trajectory at the end of the ejection phase (the case of ejection of a satellite at the perigee of the orbit.)

TABLE 18

Influence of Ejection Errors on the Perigee and Apogee of the Orbit

| Initial orbital parameters: | | | | | |
|---|---|---|---|---|---|
| height of perigee, km .:. | 250 | 250 | 250 | 250 | 250 |
| height of apogee, km .... | 800 | 1500 | 5000 | 15000 | 30000 |
| Change in orbital parameters with ejection-velocity errors of ± 10 m/sec: | | | | | |
| Change in height of perigee, km.. | 0 | 0 | 0 | 0 | 0 |
| Change in height of apogee, km... | ± 39 | ± 40 | ± 73 | ±190 | ± 472 |
| Change in orbital parameters with ejection-angle errors of ± 1°: | | | | | |
| Change in height of perigee, km.. | - 25 | - 13 | -4,8 | -2,8 | - 2,5 |
| Change in height of apogee, km... | + 25 | + 13 | +4,8 | +2,8 | + 2,5 |

From the table it is evident that a velocity error causes a corresponding change in the height of the apogee and the circling time of the satellite, without affecting the height of the perigee. For low orbits, the change in height of the apogee is only weakly dependent on the initial values of the orbital paramenters. However, as the orbital height increases, the influence

of velocity errors on the height of the apogee and on the circling time of the satellite increases noticeably.

An error in the ejection angle always results in a decrease of the perigee height and a corresponding increase of the apogee height. Since satellite deceleration occurs mainly near the perigee, this results in decreased satellite lifetime.

For an orbit with an apogee height of 800 km and a perigee height of 250 km, a 1° error in the angle at the end of the ejection phase decreases the perigee by 25 km, causing a two-fold decrease in the lifetime of the satellite.

The influence of ejection angle error on the height of the perigee decreases with increasing excentricity of the orbit, i.e., with increasing height of the apogee and unchanged perigee height. For an orbit with a perigee of 250 km and an apogee of 1500 km, the perigee decreases by 13 km due to the same error in the ejection angle as above. There is only a 1.5-fold decrease in the lifetime of the satellite due to this ejection angle error. Consequently, for more elongated orbits the decrease in lifetime of the satellite due to ejection errors will be less than for orbit close to circular.

Angle errors do not affect the circling period since in this case the value of the major axis of the elliptical orbit remains constant.

Let us now show the influence of errors at the end of the ejection phase on the orbits of space vehicles for lunar flights. Let us treat first the most favorable case, when the orbit of the space vehicle lies in the plane of the moon's orbit. As has been mentioned, this will be the case when a rocket is launched from the equatorial regions.

Let us assume that the calculated orbit passes through the center of the moon. Figure 22 gives the values of maximum velocity $\delta V_0$ and angle $\delta\vartheta_0$ errors (corresponding to orbital inclination from the center to the edge of the moon*)

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* See Uspekhi Fizicheskikh Nauk (Progress of Physical Sciences),43, No.1a,101,1957.

From these graphs it follows that the values of the maximum errors $\delta V_0$ and $\delta\vartheta_0$ change substantially depending on the excess velocity at the end of the ejection phase with respect to local parabolic velocity $\delta V_0 = V_0 - V_{par}$.

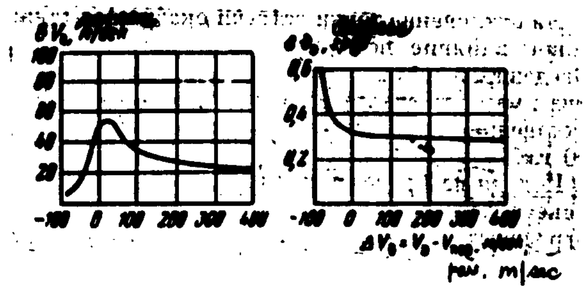

Figure 22. Values of the maximum velocity $\delta V_0$ and angle $\delta\vartheta$ errors when landing on the moon, for the case when the orbit of the space vehicle lies in the moon's orbital plane.

For velocities which are 50-60 m/sec less than parabolic, i.e., for elliptical orbits, $\delta V_0 \approx 10$ m/sec and $\delta\vartheta_0 = 0.4°$. For velocities greater than parabolic, hyperbolic orbits, the maximum velocity error increases to 20 - 40 m/sec, while the maximum angle error decreases to 0.3°. Considering, however, that orbital inclinations are determined by the combined influence of velocity and angle errors, and also bearing in mind that there are other errors which lead to orbital inclinations, we can consider that for hyperbolic orbits in the moon's orbital plane the errors at the end of the ejection phase should not exceed the following values: velocity error: 10-20 m/sec, angle error: 0.15-0.20°.

When a rocket is launched toward the moon from the middle latitudes, e.g., from the USSR, the requirements in the accuracy of the motion parameters necessary for landing on the moon are considerably increased.

For the hyperbolic orbit of the second Soviet space ship a velocity error of 1 m/sec would have resulted in a 250-km deviation in the point of impact on the moon. A deviation of the velocity vector from its calculated direction

by one angular minute would cause a 200-km shift in the impact point.

Deviations in the impact point are also noticeably affected by errors in the coordinates at the end of the ejection phase and by errors in the launch time. When the launch time deviates by 10 seconds from the calculated time there is a deviation in the impact point on the moon's surface of the order of 200 km.

From these data we can conclude that when a rocket is launched from the USSR toward the moon, the velocity error at the end of the ejection phase should not exceed several meters per second, while the velocity vector should not deviate by more than 0.1° from its calculated direction.

Even greater influence is exerted by orbital ejection errors when launching space vehicles to other planets. For a flight to Mars on an elliptical orbit which assures approach to Mars at its aphelion, a velocity error at the end of the ejection phase of 1 m/sec causes a deviation of the order of 30,000 km in the orbit of the vehicle near Mars. Therefore, for such flights the movement of the vehicle should be corrected in flight.

## Space Rockets and Carrier Rockets for Artificial Satellites Ejection into Orbit

As has already been mentioned, the basic problem in launching a space vehicle is the ejection into orbit, giving it a velocity equal to or exceeding orbital velocity at the corresponding height.

The basic means for solving this problem at present is the multistage rocket with liquid-propellant engines operating on chemical fuel.* In the near future we can expect the appearance of space rockets operating on nuclear power.

The multistage (or compound) rocket, first envisioned by Tsiolkovskiy, consists of a number of connected rockets. Let us examine this, using as our example the three-stage rocket shown in Fig. 23. Each of the three rockets has its own engine and tanks for fuel and oxidizer. The first stage includes all three rockets, the second and third rockets being, as it were, the payload of the first rocket. When the engine of the first rocket burns out it is separated, and the second stage, consisting of the second and third rockets, continues the flight. When the engine of the second rocket burns out, it is also separated and the third rocket (third stage) continues the flight alone.

Thus, in a multistage rocket as the fuel is expended the individual stages drop away. Therefore its acceleration, for the same reactive force, is greater than that of a single-stage rocket with the same characteristics. As a result, the rocket acquires greater velocity.

* In certain cases a space rocket may have individual stages using solid-fuel (powder) engines.)

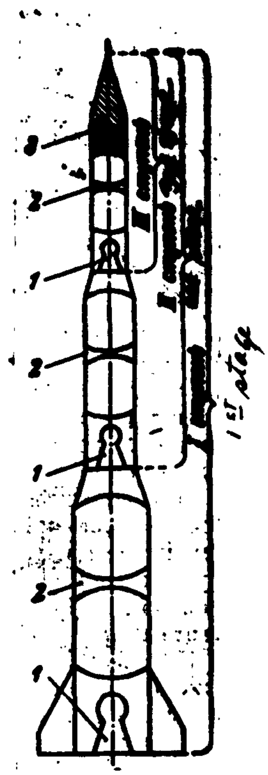


Figure 23. Diagram of a three-stage rocket.
1) engines; 2) fuel tanks; 3) payload (space vehicle)

Figure 24 is a diagram of the trajectory for ejecting a satellite into orbit. The angle of inclination of the velocity vector to the horizon at a given point of the trajectory is designated by $\vartheta$. The carrier rocket is launched vertically ($\vartheta_s = 90°$). Then, after a short vertical-ascent phase, the rocket gradually begins to turn about its transverse axis according to a set program, resulting in a curve in its motion trajectory. Be selecting a corresponding program for the turn of the rocket with time, we can obtain the required values for the height $h_o$ and angle $\vartheta_o$ at the end of the release phase.

When the satellite is ejected into orbit, $\vartheta_o = 0$ (Fig. 24). When space vehicles are launched toward the moon or other planets, and angle $\vartheta_o$ as a rule does not equal zero, but is determined by the launch condition: the date of launch, the mutual position of the planets, the geographic latitude of the launch site, etc.

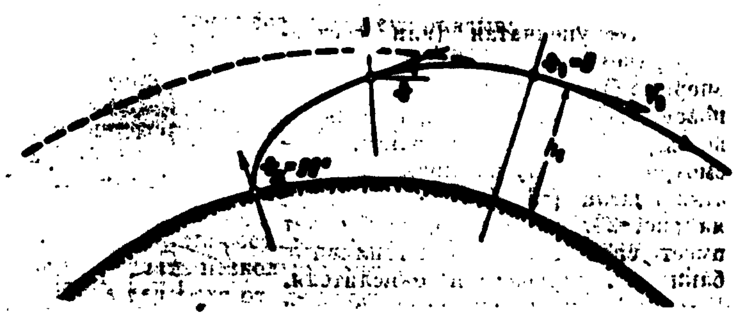

Figure 24. Diagram of the trajectory for the ejection of satellite into orbit.

Let us analyze the basic factors which determine the velocity of a multi-stage rocket at the end of the ejection phase. Let us examine the equation of motion of a rocket in projection onto the tangent to the trajectory; this equation has the form

$$\frac{G}{g_0}\cdot\frac{dV}{dt}=P-X-\frac{G}{g_0}g\sin\vartheta, \tag{1.37}$$

where G is the current weight of the rocket; dV/dt is the acceleration of the rocket; g is the acceleration due to gravity; $g_0$ is the acceleration due to gravity at the earth's surface (at sea level); P is exhaust thrust; X is the drag force; and $\vartheta$ is the angle of inclination of the velocity vector to the horizon.

The thrust of the rocket can be expressed in the form

$$P=-\frac{c}{g_0}\cdot\frac{dG}{dt}, \tag{1.38}$$

where c is the jet velocity; $c/g_0 = P_{sp}$ is the specific thrust of the engine; and $-(dG/dt)$ is the fuel consumption (change in weight of the rocket per unit time).

Then Equation (1.37) can be given in the form

$$\frac{dV}{dt}=-\frac{c}{G}\frac{dG}{dt}-\frac{X}{G}g_0-g\sin\vartheta. \tag{1.39}$$

Integrating the equation from the launch time (t = 0) to the time corresponding to the end of the ejection phase ($t = t_f$), we get the velocity of the rocket at the end of the ejection phase:

$$V_c = \sum_{1}^{m} c_i \ln\left(\frac{G_{0i}}{G_{ki}}\right) - \int_{0}^{t_k} \frac{gX}{G}\,dt - \int_{0}^{t_k} g \sin\theta\,dt \tag{1.40}$$

where m is the number of stages; and $c_i$, $G_{o_i}$, and $G_{k_i}$ are, respectively, the exhaust velocity, the initial and the final weight of the individual stages.

The first term on the right in (1.40) corresponds to the Tsiolkovskiy formula and defines the velocity of a rocket when no external forces act on it, the so-called characteristic velocity of the rocket. The second term on the right characterises the velocity lost due to predomination of drag forces, and the third term characterizes velocity lost to gravity. We have neglected, in (1.37), the losses in velocity due to non-correspondence between the direction of thrust and the velocity vector (angle of attack), since these are relatively slight.

When a space vehicle is ejected into orbit, the total velocity lost due to gravity and drag is about 2000–3000 m/sec, on the average. Therefore, to impart to a satellite a velocity of the order of 8000 m/sec the carrier rocket should have a characteristic velocity of about 10,000–11,000 m/sec, while to impart to a space vehicle a velocity of the order of 11,000 m/sec the carrier rocket should have a characteristic velocity of about 13,000–14,000 m/sec.

The characteristic velocity of a multistage rocket, as can be seen from Equation (1.40), equals the sum of the product of the exhaust velocities, in natural logarithms, and the ratios of the initial and final weights for the individual stages.

If we assume that the exhaust velocity c is identical for all stages, and the relative design weights for each stage $a_i = G_{d_i}/G_{o_i}$ are equal ($a_1 = a_2 = \ldots = a_{m-1} = a_m = a$), we can show that in the optimal case, which assures maximum characteristic velocity, the initial weights of the stages should be distributed according to the law of geometric progression:

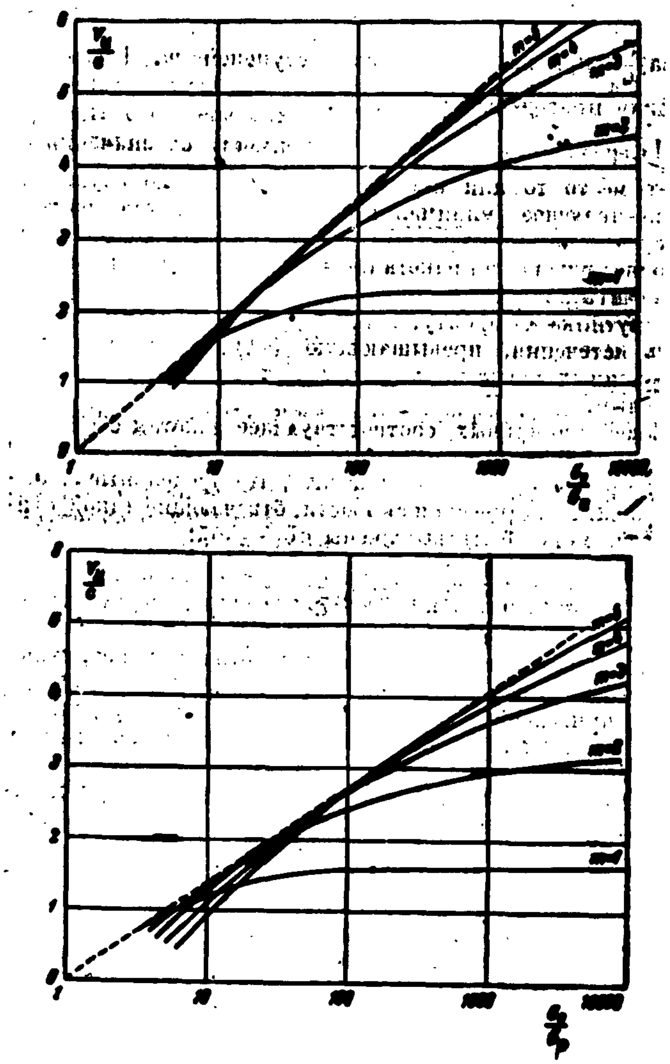

**Fig. 25. Relative characteristic velocity of rockets having various numbers of stages vs. the ratio of initial weight to payload weight, top: $\alpha$ = 0.10; bottom: $\alpha$ = 0.20.**

$$\frac{G_{o_1}}{G_{o_2}} = \frac{G_{o_2}}{G_{o_3}} = \ldots = \frac{G_{o_{m-1}}}{G_{o_m}} = \frac{G_{o_m}}{G_p} \tag{1.41}$$

or

$$\frac{G_{o_i}}{G_{o_{i+1}}} = \left(\frac{G_{o_1}}{G_p}\right)^{1/m}$$

where $G_{o_1}$, $G_{o_2}$, ..., $G_{o_m}$ are the initial weights of the stages; $G_p$ is the weight of the payload (space vehicle). The initial weight of the first stage equals the total initial weight of the rocket, $G_{o_1} = G_o$.

The characteristic velocity of such a multistage rocket can be expressed by the formula

$$V_u = mc \ln \frac{1}{\left[\alpha + \left(\frac{G_p}{G_0}\right)^{1/m}\right]}. \tag{1.42}$$

The connection between the parameters in this formula is given conveniently in the form of a graph of the relative characteristic velocity $V_u/c$ vs. the ratio of the initial weight of the rocket to the weight of the payload $G_o/G_p$ for various numbers of stages m. Figure 25 shows such graphs.

From these it is evident that, depending on the value $V_u/c$, there is a certain number of stages which assures a minimum ratio between rocket weight and payload weight. In addition, it also follows from the graphs that a single-stage rocket ($m = 1$) cannot have the velocity necessary for ejecting a satellite into orbit if it does not have an exhaust velocity greater than 5000 m/sec, which is impossible for modern chemical-fuel rockets.

The family of curves which corresponds to rockets with various number of stages has an envelope (dashed line in Fig. 25). We can show that the maximum characteristic velocity for such an envelope can be expressed by the formula

$$V_u = Kc \ln \frac{G_0}{G_p}, \tag{1.43}$$

where the value of K depends on the relative design weight (Fig. 26).

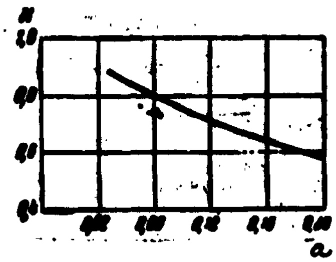

Fig. 26. Value of the coefficient K vs. a.

From this formula it is evident that the maximum velocity of a rocket with optimum parameter selection depends on the ratio of its initial weight to payload weight, the jet velocity, and the relative design weight. In this case, the initial weight of the rocket which assures ejection of a space vehicle of a certain weight into orbit, as can be seen from Fig. 25, varies within broad limits, depending on the specific values of the exhaust velocity and the relative design weight.

When a satellite is ejected into an orbit having a given perigee and apogee height, its motion in orbit can be begun from any point on the orbit, generally speaking. It is merely necessary to carry the satellite to a height corresponding to the given point on the orbit and impart to it the necessary velocity in the direction of the tangent to the orbit at this point. The higher the point of the orbit, the less velocity must be imparted to the satellite. Minimum velocity must be imparted to the satellite if its orbital motion begins at the apogee.

However, from what has been said we cannot draw the conclusion that it is expedient to eject a satellite into the apogee of an orbit. Analysis of the problem has shown that the additional expenditure of energy necessary to lift the satellite to a great height exceeds the energy gained by virtue of the fact that less velocity must be imparted. Therefore, energywise, it is best to eject a satellite into orbit near its perigee.

When the perigee of the given orbit is relatively low (of the order of

hundreds of kilometers) the satellite can be ejected immediately at the end of the powered phase of the trajectory of the carrier rocket, as shown in Fig. 24. As the ejection height increases, there is also increased power expenditure due to the effect of gravity in the ejection phase.

For high orbits, when the perigee can be at several thousands of kilometers, such an ejection method cannot generally be used because of limitations of the powered phase of the trajectory. In this case, the satellite must first be ejected into a certain transitory orbit having a relatively low perigee. At a certain point on this transitory orbit the carrier rocket should impart to the satellite an additional velocity to assure its transition to the given orbit. Thus, in this case the ejection trajectory will consist of two powered phases, separated by an inertia-flight phase.

Analysis of such an ejection method has shown that energywise, the optimum ejection of a satellite is that along a semielliptical transitory orbit whose perigee is as low as possible and whose apogee coincides with the perigee of the final orbit (Fig. 27).

The angle of inclination of the velocity vector at the end of the ejection phase into the transitory orbit, at point P', is zero ($\theta_0 = 0$). Secondary cut-in of the rocket engine occurs at point A'. The thrust direction should coincide with the tangent to the trajectory at this point.

Table 19 gives the calculated velocities which must be imparted to the satellite to eject it into circular orbits of various heights $h_{cir}$ using this method. In the calculations it was assumed that the perigee of the transitional orbit is 200 km.

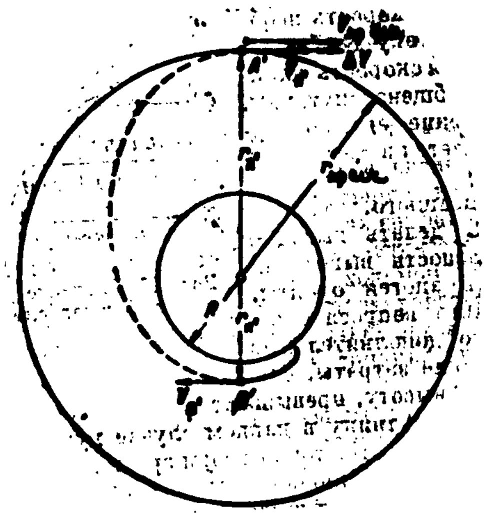

Fig. 27. Diagram of the ejection of a satellite into a circular orbit from a semi-elliptical transitory orbit. R – earth's radius; $r_{cir}$ – radius of the given circular orbit; P' – perigee of the transitory orbit; A' – apogee of the transitory orbit; $V_{P'}$ – velocity at the perigee of the transitory orbit; $V_{A'}$ – velocity at the apogee of the transitory orbit; $\Delta V$ – additional velocity needed for the space vehicle to enter the circular orbit; $V_{cir}$ – circular velocity for the given orbit.

TABLE 19

Velocity for Ejecting a Satellite from a Semi-Elliptical Transitory Orbit

| Height of the given circular orbit, km | Velocity at the end of the ejection phase into the transitory ellipse, $V_{P'}$, m/sec | Additional velocity for transition to the given orbit, $\Delta V$, m/sec | Total velocity, $V_{\Sigma}$, m/sec |
|---|---|---|---|
| 1000 | 8009 | 214 | 8223 |
| 5000 | 8769 | 854 | 9623 |
| 25000 | 10016 | 1467 | [illegible] |
| 50000 | 10424 | 1444 | [illegible] |
| 75000 | 10595 | 1366 | [illegible] |
| 100000 | 10680 | 1276 | [illegible] |
| 125000 | 10749 | 1204 | [illegible] |
| 150000 | 10790 | 1143 | [illegible] |
| 200000 | 10844 | 1045 | [illegible] |
| | 11015 | 0 | [illegible] |

From the data in the table it follows that the total velocity required to eject a satellite first increases with increased height of the given orbit, and then decreases somewhat, tending toward the limit (at $h_{cir} = \infty$) at parabolic

velocity. The total velocity is maximum when the height of the circular orbit is approximately 100,000 km. For this case, the total velocity is 8.5% greater than the parabolic velocity at the perigee of the transitory orbit (200 km). Here we have a paradoxical phenomenon in which less total velocity it required to launch a satellite to greater heights.

When a space vehicle is launched in an easterly direction the velocity imparted to it by the rocket is combined with the velocity of the earth's surface in its daily motion.

The increase in velocity due to the earth's rotation is a function of the inclination of the orbit, and can be expressed, approximately, by the formula

$$\Delta V_{rot} = \omega(R + h_0)\cos i, \tag{1.44}$$

where R is the earth's radius; $h_0$ is the ejection height; $\omega$ is the angular velocity of the earth's rotation about its axis; and $i$ is the orbital inclination.

With decreasing orbital inclination, $\Delta V_{rot}$ increases, reaching about 460 m/sec for an equatorial orbit. For polar orbits, $\Delta V_{rot}$ is zero, while for orbits with an inclination of 65°, corresponding to the orbits of the first Soviet satellites, it is about 200 m/sec.

Liquid-propellant chemical-fuel rockets have been widely developed and are now quite perfected.

The development of nuclear technology has made it possible to discuss the creation, in the near future, of nuclear-powered space vehicles. The thermal energy in such engines, developed by some sort of nuclear reactor, will be used to heat some working liquid (hydrogen, ammonia, water), converting it to a high-temperature gas which will flow from a nozzle, creating exhaust thrust.

The basic advantage of such engines over chemical-fuel engines is the possibility of producing higher jet velocities (specific thrust), since the energy imparted to the working liquid in the reactor can considerably exceed

the energy released during the combustion of even the most highly effective chemical fuels.

The exhaust velocities of such engines will be limited only by the maximum temperature which the reactor and nozzle materials can sustain.

The motion characteristics of such rockets will differ little from those of chemical-fuel rockets.

A particular class of space rockets are those space vehicles with so-called electro-reaction engines. One of these is the ion engine, in which reaction power is created by the exhaust of a stream of ions accelerated to very high velocities by means of an electrostatic field. Elements which are easily ionized, e.g., cesium or sodium, are proposed as the working substance used to form the stream of ionized gas. The energy required to accelerate the ionized gas can be obtained from a nuclear power plant, a type of atomic power station in the space vehicle.

Figure 28 is a block-diagram of an ion engine. The working substance is in tank 1, from which it is fed by means of a suitable system 2 to the engine. After being heated to a high temperature and passing through porous wall 3, the working substance in the engine is converted into ionized gas which is then accelerated in an electrostatic field created by the system of grids 4. To avoid formation of a space charge which would prevent further exhaust, the ion flux should be neutralized after acceleration; electron emitter 5 is used for this purpose. The engine is powered by nuclear power plant 6 through transformer 7.

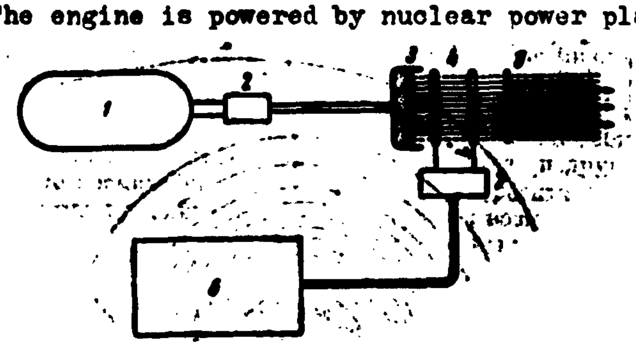


Fig. 28. Diagram of an ion engine. 1) tank with the working substance (cesium); 2) system for feeding the working substance; 3) porous wall; 4) system of grids to create an electrostatic field; 5) electron emitter; 6) nuclear power plant; 7) transformer.

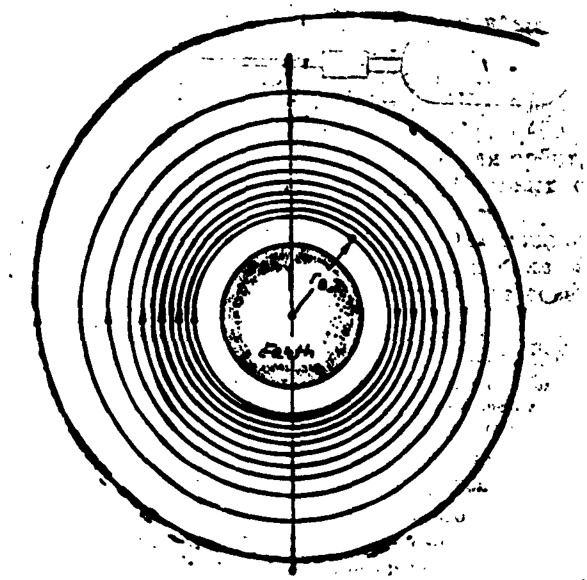

Fig. 29. Diagram of the acceleration of a
space vehicle having an ion engine

The main feature of this type of engine, which distinguishes it from other types of rockets, is the extremely high jet velocity (to 100-200 km/sec) and the very low thrust and acceleration in the powered phase (of the order of $10^{-3}$ m/sec$^2$).

The impossibility of producing high thrust from the ion engines is due to the fact that with high exhaust velocity, the power necessary for creating an exhaust jet with high thrust is excessively high. For example, to produce a thrust of 1 m at an exhaust velocity of 100 km/sec we need a power of about one million kilowatts.

Therefore, although chemical-fuel rockets can be launched independently from the surface of the earth or other planets, since the engine thrust exceeds the initial weight, space vehicles with ion engines cannot. They must begin their flight from the orbits of artificial satellites of the earth or of other planets. Their trajectory is a slowly developing spiral (Fig. 29). The engines of such vehicles can operate for many weeks.

The basic advantage of space vehicles with electro-reaction engines is the more favorable ratio of payload weight to initial weight. Therefore there is every reason to believe that in the future such vehicles will be the basic method for making flights between the orbits of artificial planet satellites.

## The Problem of Descent to the Surface of the Earth and Planets

Landing a spacecraft on the surface of the earth and planets is one of the most complex problems associated with interplanetary flight. The motion of any spacecraft relative to the earth or other celestial body takes place with a velocity equal to or greater than circular velocity. During descent this relative velocity must be reduced by some method to zero at the instant of landing. Two methods of decelerating spacecraft during descent which differ in principle are presently feasible. The first of these is based on the use of a reactive force, the second, on the use of aerodynamic forces arising during motion of the spacecraft in the atmosphere.

To accomplish the first method of descent, the spacecraft (or its descending part) must be provided with a power plant and a fuel supply to decelerate the spacecraft.

Its characteristic velocity in addition must equal the sum of the velocity of motion relative to the surface of the planet at the start of descent and the increment in velocity caused by the effect of gravitational forces during the descending phase:

$$V_{sur} = V_{rel} + \int g \sin\theta \, dt. \qquad (1.45)$$

Approximate values are given in Table 20 for the characteristic velocity when using chemical-fueled motors for descending on the surface of the earth, moon, and certain planets. This Table shows the ratio of the payload to the initial weight of the spacecraft (assuming that it has an optimal number of stages, the relative weight of the structure is a = 0.1, and the jet velocity is c = 4000 m/sec). In addition two cases are considered: descent from a parabolic orbit and descent from a circular orbit located 1000 km over the planet's surface.

We see from the Table that when using the most efficient chemical fuels, the relative weight of the payload can be: during descent on the moon, to 30-60%; on Mars, to 15-30%; to earth and Venus, less than 10%. With descent on Jupiter and Saturn the payload is practically zero. Such a result is completely natural since the problem of deceleration of the spacecraft moving with space velocity is, from an energetic point of view, equated to the problem of imparting such velocity to it.

However, in spite of the indicated shortcoming of the described method of descent, it is the only possibility when landing on celestial bodies without a sufficiently dense atmosphere and, in particular, when landing on the moon.

The second method of descent, deceleration of the spacecraft by aerodynamic forces, if the celestial body has an atmosphere. As will be shown below, the accomplishment of such a descent is most favorable under the condition that the spacecraft is preliminarily converted into an artificial satellite moving to a sufficiently low orbit close to circular.

TABLE 20
Characteristic Velocity and Relative Weight of the Payload During Descent with Use of Reactive Forces

| | Earth | Moon | Venus | Mars | Jupiter | Saturn |
|---|---|---|---|---|---|---|
| Descent from parabolic orbit | | | | | | |
| Character. vel. km/sec. | 13,0 | 2,5 | 12,0 | 5,8 | 68,5 | 41,0 |
| Rel. wt. payload | 1,5 | 30 | 1,8 | 14,3 | 0 | 0,0001 |
| Descent from circular orbit | | | | | | |
| Character. vel. km/sec | 8,1 | 1,5 | 7,4 | 3,4 | 46,0 | 27,5 |
| Rel. wt. payload | 6,7 | 61 | 8,5 | 32 | 0,00001 | 0,001 |

During descent with aerodynamic forces we must consider the two main cases:

a) Only the drag force acts on the spacecraft and its motion in this connection occurs over a ballistic trajectory;

b) In addition to the drag force, and aerodynamic life force acts of the spacecraft and it moves along a gliding trajectory.

In the first case the spacecraft, or its descending part, can be represented as an axisymmetric body moving with zero angle of attack. In the second case, it should have lifting surfaces.

We will examine the first, simplest case to begin with. The drag force acting on a body moving in the atmosphere with high velocity is determined by the formula:

$$R = C_x S_M \frac{\rho V^2}{2}, \qquad (1.46)$$

where $C_x$ - the drag coefficient,

$S_M$ - the frontal area,

p - density of the atmosphere at a given height,

V — velocity of the spacecraft's motion.

Correspondingly, the acceleration force is

$$a = \frac{R}{G} = \frac{C_x}{G/S_M} \cdot \frac{\rho V^2}{2} \qquad (1.47)$$

where $G/S_M$ is the load on the frontal area.

The magnitude of the drag acting on the spacecraft with a diameter of 1 m in relation to the height and its velocity of motion is shown in Table 21.

TABLE 21

Drag Versus Height and Velocity, tons

| [illegible] | [illegible], km | | | | | |
|---|---|---|---|---|---|---|
| | 20 | 30 | 40 | 50 | 60 | 80 |
| 1000 | 4,14 | 0,[illegible] | 0,18 | 0,045 | 0,015 | 0,[illegible] |
| 2000 | 16,55 | 3,54 | 0,70 | 0,18 | 0,06 | 0,006 |
| 3000 | 37,23 | 7,97 | 1,52 | 0,40 | 0,14 | 0,015 |
| 4000 | 66,20 | 14,15 | 2,79 | 0,71 | 0,25 | 0,027 |
| 5000 | 103,4 | 22,12 | 4,36 | 1,11 | 0,38 | 0,042 |
| 6000 | 148,9 | 31,85 | 6,28 | 1,60 | 0,55 | 0,[illegible] |
| 7000 | 202,7 | 43,36 | 8,55 | 2,18 | 0,75 | 0,[illegible] |
| 8000 | 264,8 | 56,63 | 11,16 | 2,85 | 0,97 | 0,[illegible] |

As is apparent from the data cited, the drag, and consequently the

acceleration force acting on the spaceship can reach quite large values if the motion of the spacecraft with a velocity close to space velocity occurs at low heights. Consequently, the motion of a spacecraft in the atmosphere should take place over such a trajectory at which a gradual decrease in the velocity of its motion is accomplished as the height decreases.

This requirement is satisfied by trajectories at small negative inclination angles of the velocity vector to the horizontal when entering the dense layers of the atmosphere at heights of 80–100 km. Table 22 shows the calculated data characterizing the increase in the value of the maximal acceleration forces acting on the spacecraft in relation to an increase in the angle of entrance into the dense layers of the atmosphere $\vartheta_{en}$.

TABLE 22

Increase in the Value of the Maximal Acceleration Forces with an Increase in Angle of Entrance into the Dense Layers of the Atmosphere

| Entrance angle, $\vartheta_{ent}$ | Ratio of acceleration forces to those in a trajectory corresponding to $\vartheta_{ent} = 0$ |
|---|---|
| 0 | 1,0 |
| 2,5° | 1,2 |
| 5,0° | 1,85 |
| 7,5° | 2,9 |
| 10° | 4,0 |

As we see from the Table, an increase in the entrance angle into the dense layer of the atmosphere from 0 to 5° leads to an increase in the maximum value of the acceleration forces by a factor of 2, and to 10° by about a factor of 4. With sloping descent trajectories the maximal value of the acceleration forces is about 8–10. Calculations show that the value of these overloads depends little on the load on the frontal area and drag coefficient of the spaceship. At the same time the values of these parameters affect the velocity of the spacecraft at the end of the descent phase, before its landing. The magnitude of this velocity during descent along sloping trajectories is close to the

velocity of the spacecraft's free fall in the atmosphere and is several hundreds of meters per second.

Simulateously with the effect of the aerodynamic forces on the spacecraft there takes place its intense aerodynamic heating. The kinetic energy which the spacecraft had during entrance into the atmosphere is converted to heat energy, causing an increase in the heat content and flow temperature of the air washing the spaceship. An idea of the magnitude of this energy can be obtained from the following figures. The kinetic energy per 1 kg of weight of the artificial satellite moving at a height of several hundreds of kilometers, corresponds to a thermal energy of about $2.8 \cdot 10^5$ kcal/kg. If we assume that all this heat is transmitted to the spaceship, then it is more than enough to destroy completely the spaceship regardless of its construction.

Therefore, the main problem in realising aerodynamic deceleration of a spacecraft lies in dissipating as much thermal energy as possible into the ambient atmosphere so that the spacecraft absorbs a minimum of the heat being released.

The picture of aerodynamic heating of a spacecraft in the atmosphere can be represented in the following manner (Fig. 30). In front of the moving spacecraft there occurs a compression of the gas and the so-called shock wave arises. The parameters of the gas behind the shock are sharply changed— its temperature and pressure increase, physicochemical alterations of the gas occur (dissociation, ionization, etc.). Furthermore, an increase in the gas temperature takes place in the so-called boundary layer in which there is a deceleration of the oncoming flow relative to the surface of the spacecraft. A significant amount of heat is transmitted from the heated gas washing over the spacecraft to its surface. The remaining part of the released heat is carried away by the heated gas and is dissipated in the atmosphere.

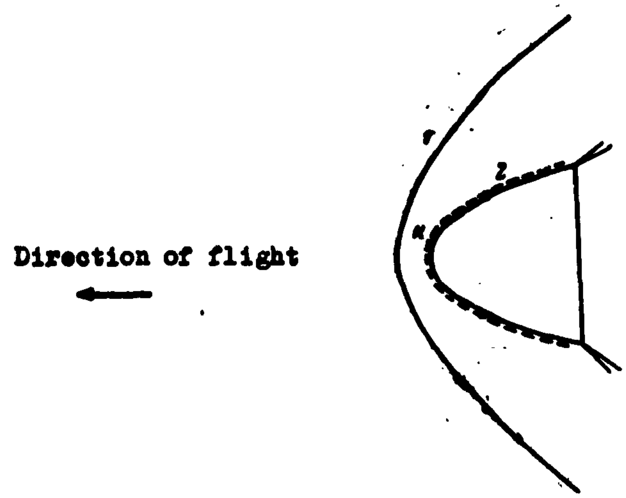


Fig. 30. Diagram of the motion of a spacecraft in the atmosphere: 1) Shock wave; 2) boundary layer; K = critical point.

The greatest thermal flux impinging on the surface of the spacecraft is close to the so-called critical point K where there is a complete deceleration of the oncoming flow. The value of the thermal fluxes arriving at the various sections of the spacecraft's surface depends on the parameters of its motion and shape. The heat being transmitted to the spaceship is partially radiated from its surface and partially proceeds to heat its skin and is transmitted inward. The surface temperature of the spacecraft can reach values at which the most refractory materials are destroyed. When the temperature exceeds the melting point of the skin material, fusion or evaporation and carry-away by the oncoming flow of the material from the surface of the spacecraft occurs. Then a part of the heat is absorbed by the processes of fusion and evaporation.

It is evident that the outer skin of a spaceship designed for landing must be made of a material having the maximum possible decomposition temperature and requiring the maximum amount of heat for fusion or evaporation in order to avoid destruction. Moreover, the design should provide for measures of heat protection which will prevent the transfer of heat into the spaceship to its equipment and crew, so that the temperature inside the spacecraft remains within the permissible limits.

The values of the thermal fluxes and the surface temperature, as well as the amount of heat transmitted to the spacecraft during the descent phase depend on the nature of its trajectory defined by the value $\vartheta_{ent}$ and on the load on the frontal area $G/S_M$.

The minimum intensity of the heat fluxes and the smallest values of the temperatures take place with the most sloping trajectories, i.e., when $\vartheta_{ent}=0$. With an increase in the angle of entrance into the atmosphere, the intensity of the thermal fluxes and the temperature values increase considerably.

When the spaceship does not have any special devices which substantially increases its resistance (frontal-area loading of the order of several hundreds of $kg/m^2$), the maximum temperature on its surface can exceed the decomposition temperature (fusion or evaporation) of presently known materials and the intensity of the thermal fluxes can reach tens of thousands $kcal/m^2 \cdot sec$.

Severe heating of the spacecraft is explained by the fact that in this case the deceleration of the spacecraft is mainly at relatively small heights (40-50 km) where the density of the atmosphere is already sufficiently great.

Let us imagine now that the spacecraft is equipped with special devices which substantially (several tens or hundreds of times) increase its frontal area and consequently the drag.

We can imagine such devices to be, for example, parachutes made of special thermostable materials which open up before the descent of the spacecraft.

In this case the intense deceleration of the spacecraft is started at large heights (70-80 km) and by the time of descent the velocity of its motion is considerably reduced. As a result the intensity of the thermal fluxes and the maximal values of the temperatures prove to be significantly smaller than in the preceding case. It is necessary to note, however, that the creation of strong braking devices of thermastable materials is an extremely complex

technological problem.

We will now consider the second variation of descent, with the use of lifting surfaces for developing lift forces, the so-called gliding descent. The lift force makes it possible in this case to maintain a small angle between the trajectory of the spacecraft and the local horizon, i.e., to make the descent trajectory quite sloping. As a consequence of this, deceleration of the spacecraft mainly takes place at large heights, in the rareified layers of the atmosphere and over a long time. Therefore the intensity of the termal fluxes, the maximal temperature values, and the overloads of the gliding spacecraft are considerably smaller than one whose descent is over a ballistic trajectory. For gliding spacecraft it is possible to develop a design from existing materials which will not melt. In addition, by changing the lift force we can control to some extent the descent tranjectory, thus ensuring a landing in a preassigned region.

Thus are the main advantages of a gliding descent which make it possible to consider that in the future gliding spacecraft will be the chief means of descending to the surface of the earth and other planets.

We must note, however, the considerable technical difficulties which stand in the pathway of realizing this means of descent. A gliding craft should have an aerodynamic shape for its stability and controlability in an extremely wide range of velocities, from subsonic to hypersonic. Its design should preserve the preformance on heating of the outer skin to temperatures close to 1500–2000°C. The craft should be controlled by special automatic systems. For carrying out landing at acceptable landing speeds it is necessary to introduce additional lifting surfaces which open prior to landing.

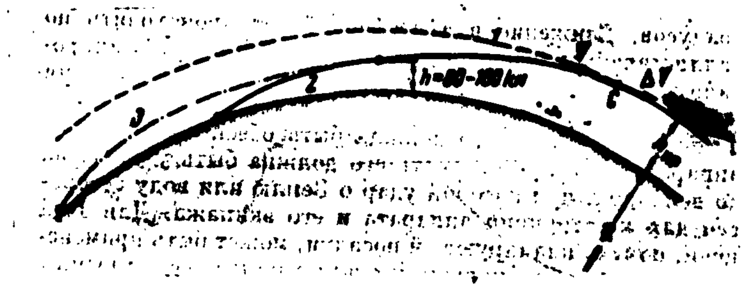


Fig. 31. Descent trajectory of a spacecraft with a circular orbit: R) earth's radius; $h_{kp}$) height of circular orbit; c) point of transition to the descent trajectory; $\Delta V$) velocity imparted to spacecraft for transition to descent trajectory; 1) section of descent trajectory lying outside dense atmospheric layers; 2) atmospheric section of trajectory with ballistic descent; 3) atmospheric section of trajectory with gliding descent.

A comparison of the two methods of descent examined above shows that the ratio of the weight of the payload to the total initial weight of the spacecraft during descent with the employment of aerodynamic forces is significantly more favorable that with descent using reactive forces. The weight of the means of heat protection, lifting surfaces, and other elements of the spacecraft is less than the weight of the fuel needed for braking the craft with jet engines.

In conclusion we will describe the process of a descent from orbit of an artificial satellite (Fig. 31). Imagine that the descent is accomplished from a circular orbit several hundreds of kilometers over the surface of the earth. For transition of the spacecraft, or its descending part, into the descent trajectory it is necessary to impart to it a certain velocity $\Delta V$ in a direction opposite to its motion in orbit. When $\Delta V$ equals 200-300 m/sec, an entrance angle of several degrees into the dense layer will be produced. The motion in the atmosphere can proceed over a ballistic trajectory 2 or over a glide trajectory 3. As a result of the aerodynamic deceleration,

the velocity of the spacecraft is reduced to several hundreds of meters per second. After this landing of the craft must be carried out, for which its velocity should be reduced to a value which will assure a safe landing of the craft and crew on the earth or water. For this purpose, besides a gliding landing, it is possible to use parachutes or retro-engines for landing.

FIRST LINE OF TEXT

## Artificial Satellites and the Problem of Interplanetary Flights

For the first time the question of the possibility of sending a space-ship beyond the limits of the earth's atmosphere was theoretically solved in the beginning of the 20th century by the outstanding Russian scientist K. E. Tsiolkovskiy, who proved that the medium for space flight should be the rocket. K. E. Tsiolkovskiy worked out a number of problems on interplanetary flight, and was the first one to suggest the principle of the rocket working on liquid fuel and laid the scientific foundation for the possibility of attaining celestial velocities with the aid of compound rockets. He is rightfully called the father of astronautics.

At the time when K. E. Tsiolkovskiy began his activity, in the beginning of the 20th century, there existed no real technological facilities for accomplishing flights out into space. However, he firmly believed in the power of the human mind. "Humanity will not remain only on the earth," he wrote, "but in striving for light and space, will at first penetrate timidly beyond the limits of the atmosphere, and then conquer all the space around the sun."

At the present time we are witnesses to a decisive step toward the accomplishment of this grandiose task.

The creation of the first artificial satellites and space rockets should be considered as a decisive practical step on the path to the accomplishment of interplanetary flights for a number of reasons.

First, the launching of artificial satellites and space rockets marks the attainment of a level of rocket technology at which it becomes possible to attain the velocities of celestial bodies, such as are necessary to accomplish flights beyond the limits of the earth's atmosphere.

Secondly, the creation of successively more perfected space equipment will make it possible to solve all basic problems in a practical way that are

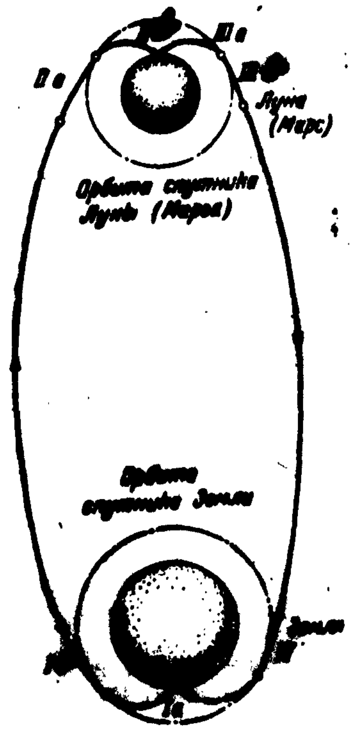


Fig. 50. Diagram of interplanetary flight:

Parts of flight: Ia--earth satellite leaving for orbit; Ib--flying away from orbit of earth satellite; IIa--transfer to orbit of moon (Mars) satellite; IIb--landing on moon (Mars); IIIa--leaving for orbit of moon (Mars) satellite; IIIb--flying away from orbit of moon (Mars); IV--transfer to orbit of earth satellite.

connected with an extended stay of an interplanetary ship with people aboard out in space. Together with this fact, the solution, for example, of such a problem as the releasing of people and equipment from artificial satellites, will prove to be at the same time the solution of the problem of bringing back interplanetary travelers from a space flight.

And finally, in accordance with modern concepts, artificial satellites of the earth and planets are necessary as intermediate stations in accomplishing interplanetary flights.

Let us dwell in detail on this question. An analysis of the prospects for interplanetary flights leads most authors to conclude that it is impossible to accomplish a flight even to the nearest celestial bodies (Mars, Venus) with the aid of a single rocket for a spaceship setting out from the earth.

The basic difficulties in accomplishing interplanetary flights can be illustrated by examples of flights to the moon and to Mars. A diagram of such flights is given in Fig. 50. The orienting values for the speeds which the rocket should attain on the different laps of its movement, and also the value for the total, so-called characteristic velocity,

are given in Table 25.

Table 25

Velocity of Rocket's Movement in Flight to Moon and Mars, km/Sec

| Required Velocity | Flight earth—moon—earth | Flight earth—Mars—earth |
|---|---|---|
| In leaving the earth: | | |
| a) for coming onto the orbit of the earth's satellite | 10.0* | 10.0* |
| b) for flying away from the orbit of the earth's satellite | 3.0 | 3.4 |
| On landing on the planet: | | |
| a) for transfer onto the orbit of the planet's satellite | 0.7 | 2.0 |
| b) for landing on the surface of the planet | 2.0* | 4.5* |
| In flying away from the planet: | | |
| a) for coming onto the orbit of the planet's satellite | 2.0* | 4.5* |
| b) for flying away from the orbit of the planet's satellite | 0.7 | 2.0 |
| In returning to the earth | 3.0** | 3.4** |
| Total characteristic velocity | 21.4 | 29.8 |

*The values presented for the velocity exceed the velocity of the movement of the satellites by the value for the losses caused by the force of gravity on the lap of the departing flight or the landing.

**Here one has in mind the changes in the velocity which assure the transition onto a circular orbit located at an altitude of some hundreds of kilometers above the surface of the earth. Further launching can be accomplished without

the consumption of fuel by taking advantage of aerodynamic forces.

From the table it is seen that the characteristic velocity in a flight to the moon amounts to more than 21 km/sec, and in the case of a flight to Mars to about 30 km/sec.

An analysis of the possibilities of rocket technology indicates that by making use of the best chemical fuels, or the methods of the use of nuclear energy known at the present time, rockets which possess the characteristic velocities shown should have an initial weight that would exceed many thousands of tons.

The creation of rockets of such great weight lies beyond the limits of the technological possibilities of the near future. Therefore the accomplishment of flights to other planets by such a very simple scheme, which assumes that the spaceship starts directly from the surface of the earth, apparently is not within the realm of possibility , at least until such a time as there shall be found new methods of obtaining reactive force which are different from those used at the present time.

However, there is another plan for accomplishing interplanetary flights-- with the use of artificial satellites of the earth and planets as intermediate stations. The idea of using artificial satellites in the accomplishment of interplanetary flights was expressed by K. E. Tsiolkovskiy and developed in the work of a number of his successors.

One of the possible variants of interplanetary flight can be presented in this case in the following way:

a) One creates an artificial satellite moving in some orbit around the earth at a sufficiently great altitude. Through successive trips of some rockets one accomplishes the transportation to the satellite of supplies of fuel and construction elements for building a space-ship. The assembling of the spaceship from separate elements is accomplished on the orbit of the satellite.

b) One accomplishes the flight from the orbit of the satellite of the earth to the planet which is the goal of the journey. The spaceship is converted into an artificial satellite of this planet.

c) With the creation in this way of an artifical satellite of the planet one accomplishes the flight to its surface and back again. For this flight one uses a part (one of the phases) of the spaceship. Another part of it, assuring the return later to the earth, continues in the meantime to move in the orbit around the planet.

d) The flight is made from this orbit to the orbit of the artificial satellite of the earth.

e) The passengers are released with the necessary equipment from the orbit of the earth's satellite onto its surface. This lap of the flight can be accomplished almost without the expenditure of fuel, basically by using aerodynamic forces.

The apparent advantages of such a plant of interplanetary flight are, first the possibility of accumulating on the artificial satellite of the earth considerable supplies of fuels and materials for assembling a space-ship of of sufficiently great dimensions and weight. When this is done each of the rockets accomplishing the transportation of the necessary materials onto the orbit of the artificial satellite does not need to have too great an initial weight (for example, of the order of several hundreds of tons).

In the second place a considerable reduction in the weight of that part of the space-ship which is to accomplish the landing of the cabin with the passengers on the planet which is the destination of the flight and the departure from it is attained. In accomplishing the flight by the first scheme the weight of this part of the spaceship would be several times as great, so that the useful load, outside of the cabin and passengers, would have to include also those phases of the rocket necessary for the final return to the earth.

FIRST LINE One particular is that a space ship accomplishing a flight between the orbits of the artificial satellites (so-called orbital ship) can use engines that have a thrust considerably less than its weight. This circumstances opens up the real prospects for the use in orbital ship of electroreactive engines, and this makes possible a considerable increase in the useful load.

For further visual illustration of the advantages of accomplishing the interplanetary flights by the second scheme we present comparative evaluations of the basic characteristics of the rockets for the methods of flight under consideration. For this purpose we make use of the formula (1.43) which determines the ratio between the characteristic velocity of the rocket and the relationship of its weight to the weight of the useful load with the optimum figure for the phases.

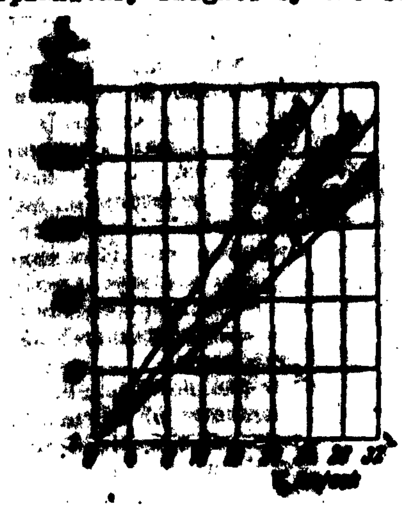

Fig. 51. Ratio of the initial weight of the rocket to the weight of the useful load as depends on the characteristic velocity.

Let us assume that the velocity at the expiration of the rocket's reactive exhaust c amounts to 4000–5000 m/sec, but the relative weight of the construction $\alpha = 0.1$, which corresponds to a very high stage of perfection in the construction of the rocket.

The ratio of the initial weight of the compound rocket to the weight of the useful lead as depends on the characteristic velocity in this case can be determined by the graph presented in Fig. 51.

On the basis of this graph one can establish that with a weight of useful load (cabin with passengers and necessary equipment) of $G_{\Pi} = 10$ tons and $c = 4000$ m/sec, the initial weight of the rocket with the aid of which it is pos-

sible to accomplish the flight to the moon by the first scheme, should amount to about 10,000 tons. For a flight to Mars under the same conditions one would need a rocket with a weight of more than 180,000 tons. With $\underline{c}$ =5000 m/sec the initial velocities would amount to: for the flight to the moon, about 3000 tons and for the flight to Mars, about 25,000 tons.

Let us consider now what characteristics, under such conditions, rockets should have in accomplishing a flight by the second method. The basic characteristics of such rockets on the assumption of using engines with a final velocity of $\underline{c}$ = 4000 m/sec with relative weight of the stages of the rockets $\alpha$ = 0.1 are shown in Table 26. With this the ratio of the initial weight of the rockets, assuring the attaining of the necessary velocities on the separate laps of the flight, to the weight of their useful load, is also determinable from the graph shown in Fig. 51. From the data given in Table 26 one sees that, in flights to the moon and Mars, the total weight of the fuel and the elements of the construction in passing from the first to the second scheme for accomplishing the flight is reduced respectively by factors of 2 and 4.5. With the second scheme the first lap of the flight can be realized through the launching of a number of freight-carrying rockets with a weight of some hundreds of tons each, whereas with the first scheme of flight one has to create airships capable of starting from the surface of the earth and possessing an initial weight of thousands and hundreds of thousands of tons.

The data presented clearly show the advantages obtained by using artificial satellites in interplanetary flights.

It is not to be denied that the accomplishing of interplanetary flights with the use of artificial satellites requires the solutions of complicated technical problems. Among such problems in the first place one finds: the precise launching of a great number of transporting rockets to the orbit of the artificial-satellite station, their approach and the assembling of a

Table 26

**Characteristics of the Rockets for Accomplishing the Interplanetary Flights with the Use of Artificial Satellites**

| Basic Characteristics | Flight earth—moon—earth | Flight earth—Mars—earth |
|---|---|---|
| Rockets for delivering the parts of the spaceship to the orbit of the artificial satellite of the earth (flight lap Ia)* | | |
| characteristic velocity, km/sec | 10 | 10 |
| initial weight (total), tons | 5,540 | 38,400 |
| overall weight of useful load, tons | 205 | 1,430 |
| Rocket for passing to the orbit of the artificial satellite of the planet (flight laps Ib and IIa)* | | |
| characteristic velocity, km/sec | 3.7 | 5.4 |
| initial weight, tons | 205 | 1,430 |
| weight of useful load, tons | 61 | 242 |
| Rocket for descending to the surface of the planet and ascending from it (flight laps IIb and IIIa)* | | |
| characteristic velocity, km/sec | 4.0 | 9.0 |
| initial weight, tons | 37 | 193 |
| weight of useful load (cabin, crew, and equipment), tons | 10 | 10 |
| Rocket for flight to the orbit of artificial earth satellite (flight laps IIIb and IV)* | | |
| characteristic velocity, km/sec | 3.7 | 5.4 |

(continued on next page

*Parts of flight shown in Fig. 50.

| | | |
|---|---|---|
| initial weight tons | 34 | 59 |
| weight of useful load (cabin with crew and equipment | 10 | 10 |

spaceship out in space, navigation while accomplishing the flights between the orbits of the artificial satellites, etc. However, this way, at the present time, apparently proves to be the only one with real prospects for accomplishing interplanetary flights.

Looking out into the future one can picture the basic stages by which mankind will accomplish flights to other celestial bodies:

a) preparatory research of the basic problems of space flight with automatic artifical satellites of the earth; accomplishing space flights of rockets with automatically working equipment;

b) the creation of artificial satellites of the earth with people on them and permanent artificial- satellite stations; detailed working out on them of all the basic problems of space flight; solving the problems of releasing people and equipment from a satellite to the earth;

c) the accomplishment by man of flights to the moon and nearby planets without landing on their surfaces;

d) passing over to interplanetary flights; subsequent study of the different planets of the solar system by the organization of expeditions to them.

From all that has been said it is apparent how great the importance of artificial satellites is for accomplishing flights to other celestial bodies.

## Study of Interplanetary Gas

The question concerning the nature and concentration of interplanetary gas is difficult to solve in the present status of astrophysics with the aid of observations, conducted from the earth's surface. This problem, which has great significance for explaining the processes of the exchange of gas between an interplanetary medium and the surface layers of the earth's atmosphere, and for studying the conditions of propagation of the sun's corpuscular radiation, may be solved with the aid of instruments, installed on rockets, which move directly in interplanetary space.

On the basis of data from observations of polarization of zodiacal light, the study of propagation of the so-called whistling atmospherics (low-frequency electromagnetic oscillations, called electrical discharges), can be taken as the most accurate model of an interplanetary medium, the constituent parts of which are characterized by the following features:

stationary plasma with temperature $T = 10^4$ K, containing electrons and protons with energies W and velocities V:

electrons $W = 0.87$ ev, $V = 6.3 \cdot 10^7$ cm/sec

protons $W = 0.87$ ev, $V = 1.5 \cdot 10^6$ cm/sec

stationary plasma with temperature $T = 10^5$ K, containing electrons and protons with energies and velocities:

electrons $W = 0.8$ ev, $V = 2 \cdot 10^8$ cm/sec

protons $W = 8.7$ ev, $V = 4.7 \cdot 10^6$ cm/sec

sporadic corpuscular flux, containing electrons and protons with energies and velocities:

electrons $W \leqslant 25$ ev, $V \leqslant 3 \cdot 10^8$ cm/sec

protons $W \leqslant 45$ ev, $V \leqslant 3 \cdot 10^8$ cm/sec

particles of external radiation belts, with which the earth is rotated at distances of several earth radii, which primarily consist of electrons and protons having energies and velocities:

electrons $W > 200$ ev, $V > 8.4 \cdot 10^{8}$ cm/sec

protons $W > 200$ ev, $V > 2 \cdot 10^{7}$ cm/sec.

For experimental verification of our presentations concerning interplanetary gas in the region of the earth and far beyond its limits, so-called proton catchers were used on Soviet cosmic rockets.

We will stop for a description of one type of these proton, or ion catchers.

A Triple-electrode, ion catcher (fig. 61) represents an instrument, composed of a collector and two grids -- internal and external, which separate the collector from the revolving space container. A negative potential relative to the body of the container $\varphi_{K}$, is maintained at the collector; negative potential $\varphi_{g1}$, the creating field, which retards the photoelectrons, which are emitted by the collector, is maintained at the internal grid; at the external grid, potential $\varphi_{g2}$ is maintained, positive, negative or sawtooth, depending on the designation of the catcher.

A variating current, flowing in the collector's circuit of such a catcher, can determine the flow density of these or any charged particles, which hit on the collector. We can "sort" these particles according to charge sign and energies, changing the voltage on the catcher's grids (or simultaneously applying several catchers with different voltages on the grids) and taking into account the sign of the total current, established by a flux of charged particles, hitting on the catcher's collector.

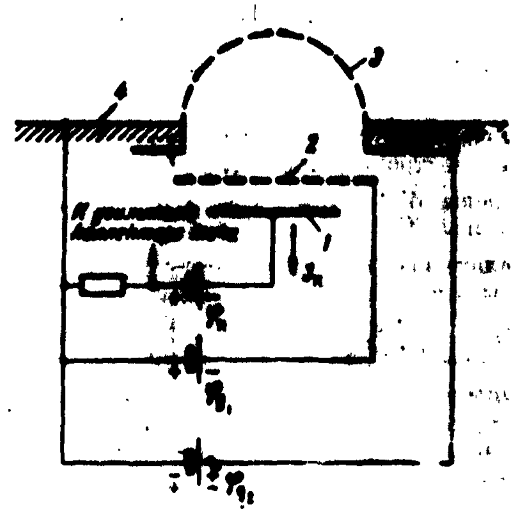

Fig. 61. Triple-electrode ion catcher; 1-- collector; 2-- internal grid; 3-- external grid; 4-- body of the container.

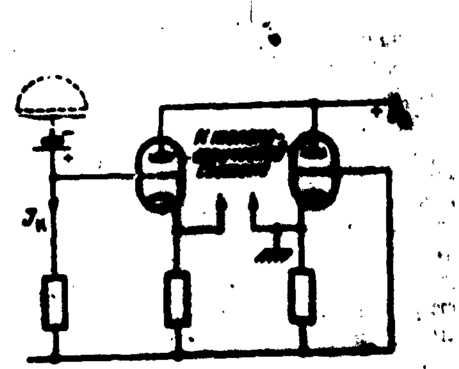

Fig. 62. Amplifying cascade.

An amplifying cascade, by means of which are produced the power of currents in the circuit of each catcher's collector, is indicated in fig. 62.



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## Biological Investigations

Biological investigations on altitude rockets and, especially on artificial earth satellites, are the most important steps in preparing for man's flight into cosmic space. To this factor is related: the effect of overloads at the rocket's start, the state of weightlessness under free flight conditions in orbit, the effect of various radiations on the living organism, the state of a highly-organized living being in a hermatic cabin, and the adaptability of a living organism to conditions, approximate to a cosmic flight. Investigations in these directions are conducted on mice, small pigs, rabbits, dogs and monkeys. Monkeys, as a rule, ascend in the rockets in a narcotic state, which significantly lowers the value of the conducted experiment. Soviet physiologists conducted a series of successful experiments on dogs at the take-offs of geophysical rockets down to an altitude of 470 km. The experimental animals were safely returned to the earth, where their condition scarcely deviated from normal. The vast material accumulated by Soviet physiologists at take-offs of altitude rockets allowed for the making of complete determined conclusions concerning the possibility of sending a living organism into cosmic space.

Biological investigations on artificial earth satellites were conducted to broaden our knowledge concerning the stay of a living organism in cosmic flight conditions. In contrast to the biological investigations on altitude rockets, the artificial satellites are used to study the effects of prolonged influence of accelerations, noise and vibrations at the launching of the satellite up to the moment of its exit to orbit and the prolonged state of weightlessness during orbital flight.

For providing an animal with all the necessary living conditions during the cosmic flight, and also for registering an animal's physiological functions,

a special apparatus is used, corresponding to high requirements in design.

Fig. 80. Physiological Apparatus Set;

1-- microphone; 2-- automatic pressure device; 3-- autonomous recorder; 4-- first amplifying and distributing block; 5-- second amplifying and distributing block; 6-- voltage conversion block; 7-- temperature elements; 8-- unit for measuring arterial pressure; 9-- breathing unit; 10-- unit for recording motion.

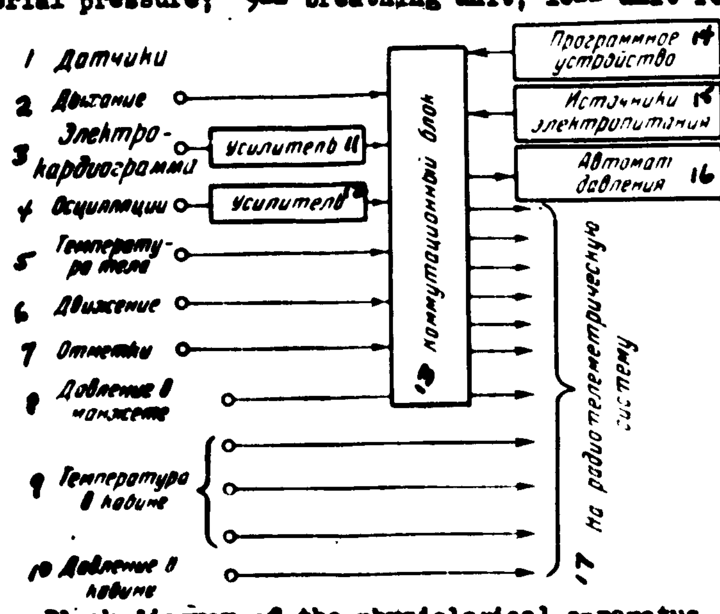


Fig. 81. Block-diagram of the physiological apparatus.



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Key to fig. 81.

1. Units
2. Respiration
3. Electrocardiograms
4. Oscillations
5. Body temperature
6. Movement
7. Marks
8. Pressure in cup
9. Cabin temperature
10. Cabin pressure
11. Amplifier
12. Amplifier
13. Commutation block
14. Program device
15. Electrical-feed sources
16. Automatic pressure device
17. To radiotelemetric system

The following apparatus is inserted in the make-up of the hermetic cabin's equipment: a regeneration device with an automatic system, a regulator of air temperature in the hermetic cabin, an automatic device for feeding and providing water for the animal, an attachment for fixing the animal's position in the cabin, and a set of physiological units together with an amplifying and commutation block and amplifiers.

With the aid of the physiological units, which are distributed on the animal (fig. 80), indices are recorded, characterizing the state of respiration and blood circulation of the animal in flight, and namely: rates of heart contraction by means of recording the biocurrents of the heart; the amounts of maximum arterial blood pressure by the oscillation method at periodic reduction of the exposed in a carotid skin shred by means of a special cup. Furthermore, the actography method if used with the motion unit for making a judgment concerning the animal's motive activity.

Biocurrents are recorded by means of silver electrodes, inserted under the animal's skin. The use of tensolytic rheostat units, applied in the form of belts on the animal's ground cell, permits the respiration rates to be recorded. The oscillation unit, which converts pulse vibrations of the carotid walls in electrical oscillations by means of a piezocrystal, records arterial pressure. The animal's movements are recorded by a potentiometric unit. A block-diagram of this apparatus is presented in fig. 81.

For providing the animal's food under weightless conditions, special food gelatinous masses are processed, containing the necessary quantity of water.

A special sanitizing unit is provided for the animal's functions.

By means of extensive training under laboratory conditions and experiments conducted many times a day, the animal is prepared for a flight on an artificial earth satellite.

Second Soviet Cosmic Rocket. First Flight To The Moon

The launching of the second Soviet cosmic rocket to the Moon was carried out on September 12,1959. The purpose of the launching was to investigate cosmic space and realization of the first flight to the Moon.The last stage of the rocket weighed, after consumption of fuel, exactly 1511 kg. Control of the rocket along the final stretch was automatic- with a special control system.

On the last stage in piggyback fasion rode the separable cosmic apparatus (capsule with scientific and radio technical equipment, the apparatus in its construc tion resembled the outfit mounted on the first Soviet cosmic rocket. Separation of the apparatus (capsule) was realized after disconnecting the power plant of the last stage. Upon separation the capsule acquired small additional velocity relative to the rocket.

The scientific instruments carried on board the cosmic apparatus, secured:

investigation of the magnetic fields of the Earth and Moon;

investigation of radiation bands around the Earth;

investigation of intensity and cosmic radiation intensity variations;

investigation of heavy particles in cosmic radiation;

investigation of gaseous component of interplanetary matter;

investigation of meteoric particles.

To transmit scientific information back to Earth and to measure the trajectory parameters the apparatus carried a radio transmitter ,operating on a frequency of 183.6 mc, as well as a radio transmitter,operating on frequencies of 39.976 and 19.993 mc. Signals of the latter came in form of pulses of variable duration from 0.2 to 0.8 sec., following at repetition frequency of 1 ± 0.15 c.

The given temperature of (20-25°C) was maintained by a thermo-control system

and proper treatment of the ship's outer surface.

In addition to the radio transmitters located in the separating cosmic apparatus, directly during the last stage was activated a radio transmitter operating on frequencies of 20.003 and 19.997 mc. With the aid of this radio transmitter, emitting signals in form of telegraph messages with a duration of from 0.8 to 1.5 sec. was carried out radio observation over the flight of the last stage and data concerning cosmic radiation intensity were transmitted. During the last stage there was also a special device for the creation of an artificial sodium comet.

Total weight of scientific and metering devices carried on board the Soviet cosmic rocket together with the power sources and cosmic apparatus was 390.2 kg.

The rocket carried banners with the state emblem of the USSR and inscription "USSR, September 1959". Safety of the banners during encounter with the Moon was provided by proper structural measures. Steps were also taken to prevent contamination of the lunar surface by terrestrial microorganisms.

The flight trajectory of the second Soviet cosmic rocket (fig.103) was selected in such a manner that during its approach to the Moon and at the moment of encounter the Moon should be over observation points situated in the USSR, near upper culmination, i.e. that its elevation above the horizon should be at maximum. The most favorable conditions for radio communication have been secured.

See Page 81a for Figure 103

Fig.103. Schematic drawing of the trajectory of second Soviet cosmic rocket
1-orbit of Moon; 2-plane of rocket trajectory; 3- scintillation of rocket; 4-plane of lunar orbit; 5- position of Moon at the moment of rocket start.

The selected trajectory, of hyperbolic type, secured the duration of the lunar

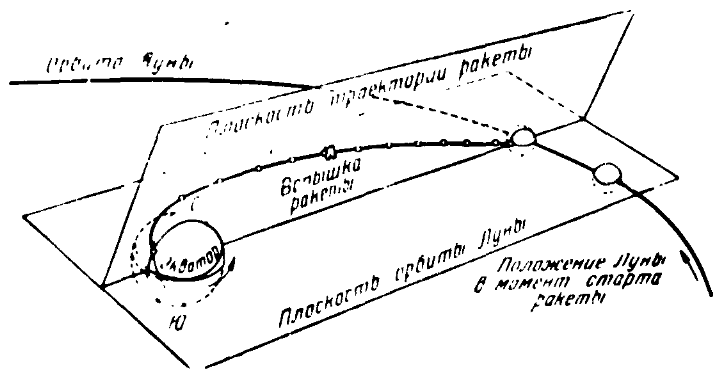


Fig. 103. Schematic drawing of the trajectory of second Soviet cosmic rocket.



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trip at about 36 hours. The velocity of motion of the rocket at the end of the separation section was somewhat higher than the local parabolic velocity.

The selection of trajectory has been preceded by a greater mathematical operation, carried out with the aid of high speed electronic computers. When making the calculations in addition to the gravitational forces of Earth and Moon it was found necessary to consider also the deviation of the terrestrial gravitational field from the central (as result of Earth's compression = shrinkage) and disturbing effect of solar gravitation. As result of the calculations was established the optimum trajectory, offering maximum value of useful load weight, and the moment of rocket blast-off has been selected.

The necessity for accurately maintaining the calculated blast-off time is determined by the circumstance, that at agiven flight direction the plane of trajectory rotates together with the Earth during its diurnal rotation around the natural axis. The blast-off of the second Soviet cosmic rocket was realized with extremely great accuracy - deviation from the given moment of time constituted about one second.

At 1500 hrs according to Moscow time on September 12, 1959 the rocket was away from the Earth by a distance of 78.5 thousand km and was over a point, situated to the north of New Guinea Island. At about 2200 hrs of this very same day the distance between rocket and Earth was 152 thousand km.

At 21 hrs 40 mi. according to Moscow time, when the rocket was observed in the Aq uarius constellation, approximately along the line, connecting the Alpha star of the Aquila constellation and alpha of Piscis Austrinus constellation, the apparatus, installed on the last stage formed an artificial sodium comet. The artificial comet became visible at 21 hrs 48 min., when the dimensions of luminous cloud of sodium vapors attained considerable magnitude. It was observed

and photographed within a period of 5-6 minutes by many observers.

On September 13, at 3 hrs 20 min acc.to Moscow time the rocket, sitauted at a distance of 200 thousand km from the Earth, disappeared from the zone of observation of the metering points situated on the territory of the USSR. At 9 hrs of September 13 it appeared from the radio horizon from eastern direction and the measuring points again began receiving scientific information and continued with the radio measurements. At that time the distance between rocket and Earth rose to 250 thousand km.

At 16 hrs.40 min.of September 13, the rocket reached the sphere of action of the Moon. It velocity of motion was about 2.3 km/sec. Further on the velocity of its motion relative to the Moon kept on increasing continuously, having reached at the moment of coming in contact with the Moon approximately 3.3 km/sec.

At 0 hrs, o2 min. 24 sec Moscow time on September 14 1959, the second Soviet cosmic rocket reached the surface of the Moon. The operation of the radio media, installed on the rocket, which functioned reliably all during the flight, was cut off at the moment it made contact with the moon.

The processing of observation data showed that the cosmic apparatus, mounted on the second Soviet cosmic rocket, descended to the surface of the Moon to the east of the YASNOST' (Brightness) sea near the Aristide crater, Archimedes crater and Avtolik crater. The selenographic latitude of the point of encounter between apparatus and the surface of the Moon, according to obtained data, equals 30°, and selenographic longitude equals zero. Deviation of the point of lunar contact from the center of the visible lunar disk is approximately 800 km. At the moment of encounter the trajectory of the apparatus was inclined toward the surface of the Moon at an angle of 60°. The processing of obtained data shows that the last stage of the rocket has also reached the surface of the Moon.

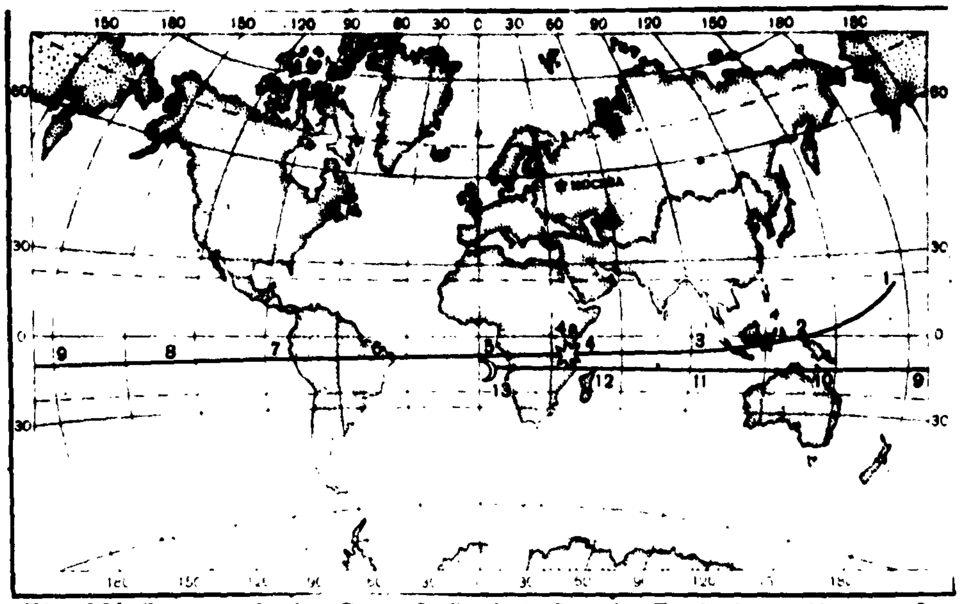

Fig.104.Route of the Second Soviet Cosmic Rocket on the surface of the Earth

Thus was realized the first cosmic flight from the Earth to another celestial body. To the surface of the Moon were delivered banners with the emblem of the USSR. The successful flight of the second Soviet cosmic rocket was one of the most important phases on the way of investigating the cosmic space and mastering of interplanetary flights.

### Third Soviet Cosmic Rocket.Automatic Interplanetary station

On October 4,1959 the USSR launched the third cosmic rocket. The purpose of the launching was the solution of many problems connected with studying cosmic space. The most important of these was the obtainment of photos of the surface on the reverse side of the Moon,inaccessible for ground observations.

To solve these problems an automatic interplanetary station was constructed, and lifted with the aid of a multistage cosmic rocket into orbit rounding the Moon. Having covered a distance of several thousand km from the Moon in accordance with calculations, the automatic station under the effect of lunar gravitation changed its heading (course). Travelling next over a new elliptical orbit,rounding the Earth the station got away from it into the apogee at a distance of about 480 thousand km Such an orbit was extremely convenient for photographing the side of the Moon invisible from the Earth and for the transmission of scientific information to the

Earth as well.

The last stage of the third Soviet cosmic rocket weighed 1553 kg (without fuel). The weight of automatic interplanetary station mounted on it was exactly 278.5 kg. In addition, the last stage of the rocket carried the metering apparatus with power sources of total weight of 156.5 kg. In this way the total weight of the useful load of the third Soviet cosmic rocket was 435 kg.

Structural improvement and high accuracy of the control system of the multi-stage cosmic rocket, used for launching the automatic interplanetary station, made it possible to lift same into orbit, practically no different from the calculated one, which guaranteed successful execution of an entire complex of scientific investigations and obtainment of the first historical photos of the reverse side of the Moon.

Arrangement of the Automatic Interplanetary Station

The automatic interplanetary station - a cosmic apparatus, equipped with a complex arrangement of different devices.

The basic systems, installed on board the interplanetary station, were:

radio-technical system warranting the measurement of station orbit parameters, transmission to Earth of TV and telemetering information, as well as transmission from Earth of commands for controlling the operation of the equipment carried on board the station;

photo-TV-system intended for photographing the Moon with subsequent automatic processing the film on board the interplanetary station and transmission of obtained image over the TV channel to Earth.

The complex of scientific devices for further investigation of cosmic space , initiated on the first Soviet cosmic rockets;

special orientation system, providing orientation of the interplanetary station relative to the Sun and Moon, necessary for photographing the invisible side of the

Moon;

power supply system for the equipment carried on board the interplanetary station ;

temperature control system.

Operation of station installations was controlled from ground points over a radio line and by autonomous programming devices carried on board the station.

Such a combined control system is most convenient for carrying out scientific experimentations and allows to obtain informat.on from any points of the orbit situated within limits of radio visibility from ground metering points.

The automatic interplanetary station had the shape of a cylinder with spherical bottoms (fig.105).Maximum lateral dimension of station - 1200 mm,length-1300 mm (antennas not considered).

The thin-walled airtight shell of the station was made of light alloy. In it were situated the entire aerial equipment of the station and the chemical power sources. On the outside was mounted a part of the scientific instruments,antennas and sections of the solar batteries.

In the upper bilge (bottom) was an illuminator with lid,opening automatically before the beginning of photographing. Under the illuminator are situated the lenses of the photo cameras and lunar orientation transmitters. On the upper and lower bottoms were also placed small illuminators for solar feelers of the orientation system. On the lower bottom are mounted power plants controlling this system (fig.106).

The radio system of the interplanetary station, as mentioned above, secured the combining of various functions into a single radio communication line. With the aid of the radio system were measured the movement parameters of the interplanetary station- distance, radial velocity and angular coordinates. In addition the radio system transmitted telemetering information,coming from scientific and control

devices, transmission of TV signals and reception from the Earth of radio-commands according to which the connection and disconnection of various instruments on board the station was executed.

All these functions in line of radio communication with the station were carried out under continuous emission of radio waves ( in contrast to the wide picked up pulsed emission). Such a combination of functions in one radio line, working under continuous emission, was realized for the first time and it gave the possibility of securing reliable radio communication all the way to maximum distances at least energy losses on board the station. The total volume of information ,transmitted over the radio line, by much exceeded the volume of information which has been transmitted from the first and second Soviet cosmic rockets.

See page 87a for Figure 105

Fig.105.Automatic Interplanetary Station ( on assembly trolley)

The radio apparatus of the interplanetary station included radio transmitters, operating on frequencies of 183.6 and 39.986 mc. The first of these served for controlling the orbital elements of the station, transmission of TV images and transmission of basic scientific information as well. A part of the scientific information was transmitted with the aid of the second transmitter. Its signals

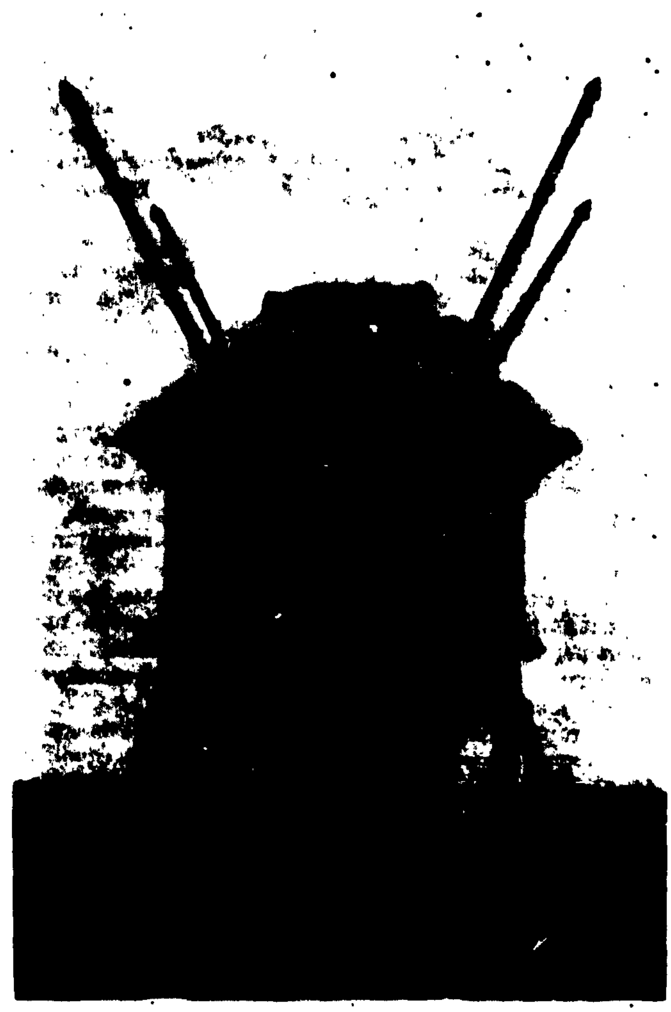

Fig. 105. Automatic Interplanetary Station

came in form of pulses (beeps) of variable duration from 0.2 to 0.8 sec.,repeating itself with a frequency of 1 ± 0.15 c.

The apparatus of the radio line was duplicated to increase reliability of communication. In case of failure of any one of the radiotechnical devices on board the station it could be replaced by a duplicating device,switched on by transmission of a proper command from the control point on the ground.

Special attention was paid to maximum weight and dimension reduction of the instruments carried on board the station. In the radio installations were widely used semiconductors, ferrites and other modern radio elements. To economize on power the power emitted by radio transmitters on board the station was fixed at several watts.

The ground radio installations,situated at metering points, had powerful radio transmitters, sensitive receivers,command and recording devices as well as antenna systems of greater effective area.

Some ideas about the difficulties connected with the task of providing reliable radio communication with interplanetary station, can be gained,when we take into consideration that the power received by the ground antenna at maximum distance between Earth and station, is approximately 100 million times smaller than the average power picked up by an ordinary TV receiver. The reception of such weak signals against the background of noise of cosmic radio radiation appears to be an extremely difficult problem and calls for the use of highly sensitive receiving devices,with low level of natural (Eigen) noise and known reduction in the rate of information transmission as well. In the radio line of the interplanetary station were used such methods of processing and transmission of signals on board the station and at ground metering points,at which the noise level was reduced to a maximum with

retention of the permissible rate of transmission.

See page 89a for Figure 106

Fig.106. General view of Automatic Interplanetary Station (Drawing) 1-illuminator for photo cameras; 2- power plant of orientation system; 3-solar feeler; 4-solar battery section; 5-louvers of thermocontrol system; 6-thermal shields; 7- antennas; 8- instruments for scientific investigations.

For photographing the Moon most appropriate was the system at which the photo cameras were aimed by rotating the entire automatic interplanetary station. The rotation and maintaining the interplanetary station on the given course was realized by an orientation system. The basic elements of this system were:optical feelers (solar and lunar), gyroscopic feelers, logical electronic units and control motors.

The orientation system wqs cut-in after the capsule drew closer to the Moon, at the moment when the station was on an approximately straight line ,connecting the Sun with the Moon. The Earth at that time was on the side from the Sun-Moon direction. The distance to the Moon at the moment of cutting-in the orientation system was, according to calculation, 60-70thousand km. It was then possible to carry out lunar orientation by illuminating the station by three bright celestial luminaries - Sun, Moon and Earth.

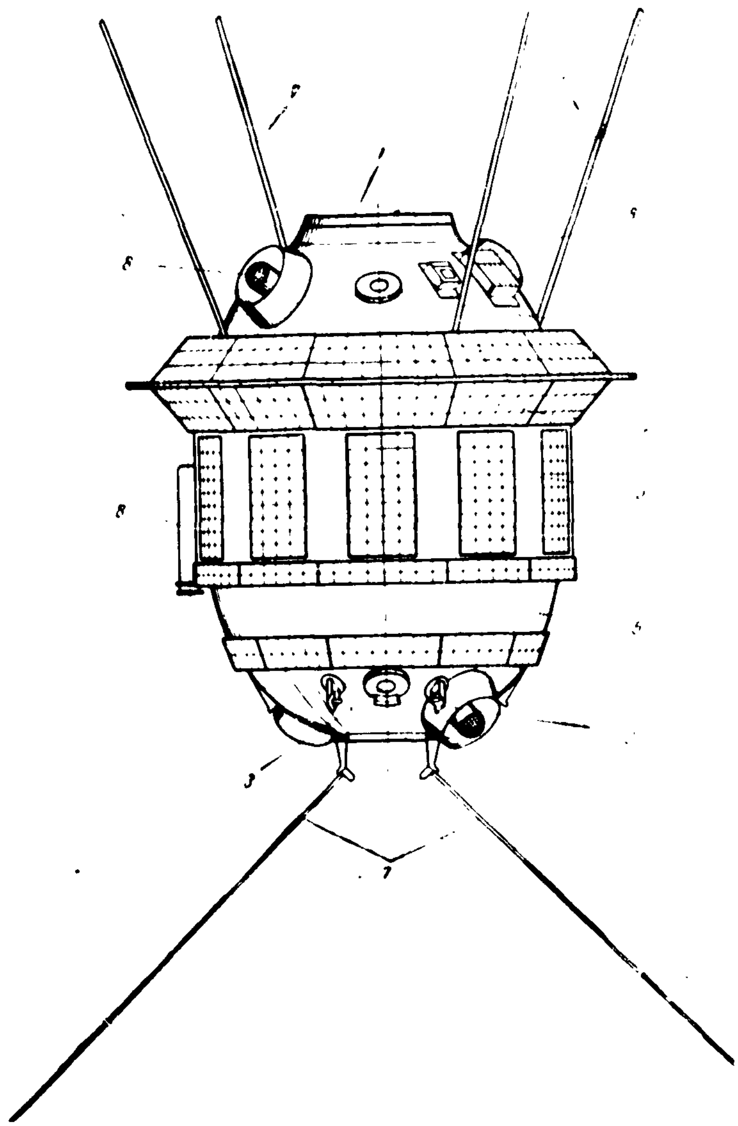

Fig. 106. General view of Automatic Interplantary Station (Drawing)

At the outset of its operation the orientation system first of all discontinued the voluntary rotation of the automatic interplanetary station around its CG, which originated at the moment of break-away of the last stage of the carrier-rocket.

After ceasing the rotations of the station with the aid of solar feelers was carried out its orientation relative to the Sun so that the lower bottom of the station was facing the Sun. At such a position of the station the optical axes of the photo cameras pointed toward the Moon.

The proper optical device, in the focus of which neither the Earth nor the Sun could appear, then cut off the feelers for orientation on the Sun, aiming the photo cameras of the station precisely toward the Moon. The signal coming from the optical device and announcing " presence " of the Moon initiated the start of automatic photographing. During the entire time of photographing the orientation system provided continuous vectoring of the automatic interplanetary station toward the Moon. Schematic drawing of the interplanetary station's orientation process is shown in fig.107.

See page 90a for Figure 107

Fig.107, Schematic of the process of orienting the interplanetary station toward the Moon:
I-VI-subsequent positions of the station. Position V-corresponds with the photographing of the Moon.

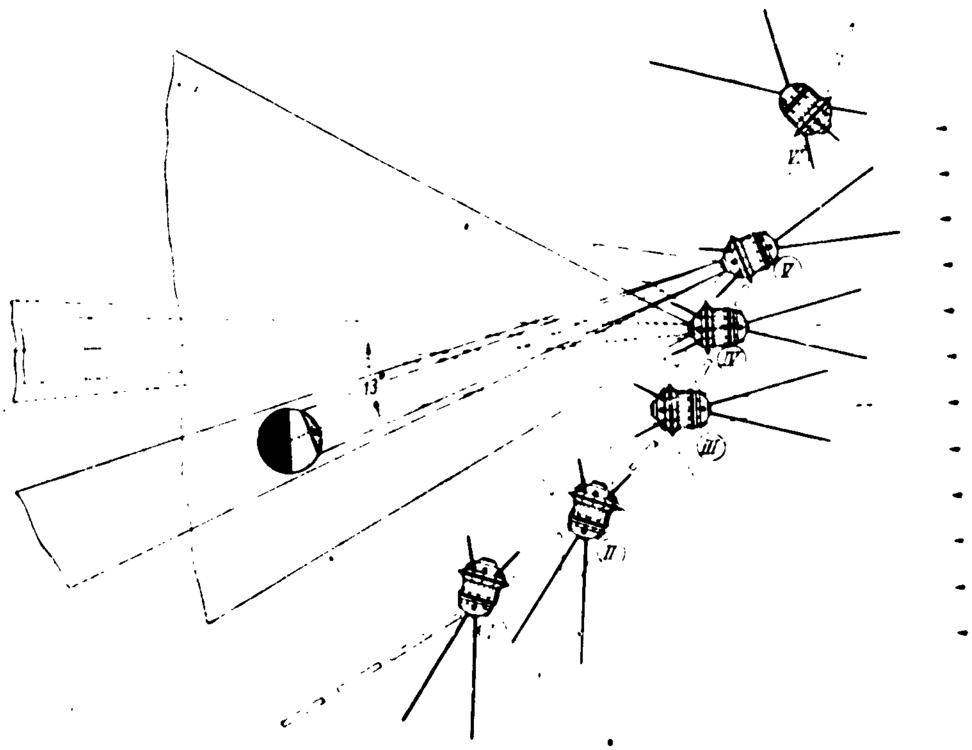

Fig. 107. Schematic of the process of orienting the interplantary station toward the Moon



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After exposing all the frames the orientation system was disconnected. At the moment of cut off it imparted to the automatic interplanetary station and ordered rotation with definite angular velocity,selected so that, on one hand,to improve the thermal process, and on the other hand, not the affect the functioning of the scientific equipment. The basic elements of the photo-TV system were: photo camera, automatic film processing arrangement, TV-apparatus.

The photo-camera was provided with two lenses with focal lengths of 200 and 500 mm and proper shutters 1:5.6 and 1:9.5.

The lens with 200 mm focal length gave the image of the lunar disk,fully fitted into the frame. The lens with 500 mm focal length gave a large scale image of a part of the lunar disk.

Photographing was done on a special 35 mm film, enduring high temperature processing. Photigraphing was done with automatic change in exposure of various frames to obtain negatives with most favorable densities and lasted for about 40 minutes, during the time of which the reverse side of the Moon was photographed repeatedly.

The entire process of photographing and film processing was executed automatically in accordance with a set program.

To prevent fogging of the film under the effect of cosmic radiation the system was provided with a special protection,selected on the basis of investigations, carried out with the aid of Soviet man-made satellites and cosmic rockets.

After the photographing has been completed the film was guided into a small size automatic processing arrangement, where the development,fixing and drying were carried out. After this the film moved to a special box and was being readied for image transmission.

Transmission of the lunar picture was done by command from the Earth.These

commands cut-in the power of the station's TV arrangement,device for stretching the film and the TV apparatus was connected to the station's transmitters.

To transform the images on the film negatives into electric signals was applied the "translucence " method, analogous to the one used by TV-center for transmission of motion picture films: small size CRT of high resolving power produced a bright luminous spot, which with the aid of an optical system was projected on a photo film. The light, having passed through the photo film, fell on a photoelectric multiplier, which converted the light signal into an electrical.

The photic spot on the screen of the CRT moved in conformity with the controlling electrical signals,produced by a special scanning arrangement. The image of the photic spot on the photo simultaneously moved across the film, from one of its edges to another, after which it rapidly restored itself into initial position to continue again its uniform movement across the film. This secured "line" scanning of the image. The photo-film in itself moved slowly past the CRT, which provided "frame" scanning.

The intensity of the light which passed from the CRT through the film to the photoelectric multiplier, is determined by the density of the negative at this point at which the photic spot is situated. When the spot travels over the negative the amperage in the photoelectric multiplier changed in conformity with the law governing the change in image density along the line; in this way at the output of the photoelectric multiplier was created an electric "image signal" repeating the law of negative density change along the line of resolution.

Amplification and formation of image signals were realized with a specially developed narrow band stabilized amplifier.

Since the average density of the negative and image contrast have not been exactly known before, the amplifier was provided with an automatic control device

providing compensation for the effect of change in average density of the negative on the output signal. Provided was also automatic brigtness control of the translucent tube, compensating for contrast changes.

On the film were preexposed test signs, parts of which were developed on the Earth, and the remaining parts developed on board the station in the process of developing exposed frames with image of the reverse side of the Moon. These signs were transmitted to Earth thus offering the possibility of checking the process of photographing, processing and transmission of image.

The process of image transmission was carried out in two ways much slower transmission at greater distances and much faster one at shorter distances, when drawing closer to the Earth.

The number of lines into which the image was broken down could vary depending upon the selected way of transmitting. The maximum number of lines reached up to 1000 per one frame.

To synchronize the transmitting and receiving scanning devices was employed a method, securing high interference resistance and operational reliability of the apparatus.

Ground reception of Moon image signals was realized on special devices recording TV images in tape, on magnetic recording devices with great stability of the rate of motion of the magnetic tape, on skiatrons (CRT with long lasting preservation of image on the screen) and on open recorders with image regidtration on electro-chemical paper. Data obtained from all forms of registartion, were used in studying the invisible part of the Moon.

With the aid of a radio-TV apparatus, carried on board the automatic interplanetary station, transmission of images was realized at various distances all the way to a distance of 470 thousand km. This gave experimental confirmation of

the possibility of transmitting in cosmic space over ultra-remote distances semitone images of high definition without substantial specific distortions in the process of radio wave propagation.

Power supply for the equipment on board the interplanetary station was furnished from autonomous units of chemical current sources and from a central power system. In this system was included a solar battery, individual sections of which were situated on the outer surface of the interplanetary station, and a chemical buffer battery. The power output of the buffer battery during operation of board equipment was compensated by the energy coming from the solar battery. Power to board equipment was delivered through transformers and stabilizers.

The automatic temperature control system maintained stable temperature in the interplanetary station, securing heat transfer, of heat emitted by instruments, through a special radiation surface into the surrounding cosmic space. To control heat transfer on the outside of the station body were installed louvers, opening the radiation surface upon a temperature rise in the station to above + 25°C.

Flight of Automatic Interplanetary Station

The operational characteristics of the orientation system and the conditions of radio communication with the automatic interplanetary station required the selection of a proper flight trajectory, satisfying a series of specific requirements.

For normal operation of the orientation system, as already mentioned, it was necessary, that at the moment it began functioning the Moon, station and Sun were situated approximately on one straight line, with the station at that time at a definite distance from the Moon.

In connection with the larger volume of information, transmitted from board the interplanetary station to the Earth, the flight trajectory had to make it possible for ground receiving points, situated on the territory of the USSR, to obtain a

maximum amount of information already during the first turn and, particularly, at short distances from the surface of the Earth.

It was also highly desirable for scientific investigation purposes to obtain a trajectory securing the movement of the interplanetary station in cosmos for quite a longer period of time.

As shown by investigations, the requirements can be best met when the gravitational effect of the Moon is used for the formation of an orbit. A considerable effect of the Moon on the movement of the interplanetary station can be attained only in the case when the attraction of the Moon is sufficiently high, i.e. when the station comes quite close to the Moon. To attain a given change in orbital characteristics the station must approach from a definite side of the Moon.

To fly around the Moon and return to Earth the velocity at the end of separation section should be somewhat lower than local parabolical velocity. In this case the flight around the Moon can be carried out at various trajectories.

If the flight trajectory passes at distances of several tens of thousands of kilometers from the Moon then the effect of the Moon is relatively low and the movement of the station relative to the Earth will follow a trajectory close to an ellipse with focus in the center of the Earth. But such trajectories for distant flight around the Moon have a number of serious shortcomings. First of all, when flying at greater distances from the Moon it becomes impossible to directly investigate cosmic space in surroundings near it. On the other hand, when launching a rocket from the northern hemisphere of the Earth, the return to Earth is from the side of the southern hemisphere, which hampers the observations and reception of scientific information by stations situated in the northern hemisphere. Movement near the Earth during return is outside of the visibility ranges of the northern hemisphere, and that is why radio communication with station travelling near the

Earth cannot be brought into realization. And finally, thirdly, When coming back to Earth over such a trajectory the rocket enters the dense layers of the atmosphere and burns up. It flight is thus terminated after the first turn.

Using, for the formation of an interplanetary station orbit, the directed effect of lunar gravitation, when the station passes close to the Moon, will allow to attain an orbit, void of shortcomings inherent of trajectories for distant flights.

The flight trajectory of the automatic interplanetary station passed at a distance of 7900 km from the center of the Moon and was selected with such consideration that at the moment of maximum nearness (approach) the station would be to the south of the Moon. Because of lunar gravitation the trajectory of the automatic station in conformity with calculation would bear northward. This deviation was so essential that the return to Earth took place from the side of the northern hemisphere. When the station came close to the Moon the maximum altitude of the station above the horizon for observation points, situated in the northern hemisphere increased from day to day (diurnal period to diurnal period). There was also a corresponding increase in the time intervals, during which it became possible to attain direct communication with the station. When approaching the Earth the automatic station could be observed in the northern hemisphere as a nonsetting star.

During return to Earth during first turn the station did not enter the atmosphere and did not disintegrate, but travelled at a distance of 47.5 thousand km from the center of the Earth, moving along an extended orbit, close in form to elliptical. Maximum distance between station and Earth was 480 thousand km (fig.108).

Flight of the interplanetary station close to Earth during the first turns took place at greater distances from its surface, where deceleration on account of atmospheric resistance was practically nil. If the movement would have taken place only under the effect of the gravitational force of the Earth, the

automatic station would have become a satellite of the Earth with unlimitedly greater life expectancy.

The fact is that the time of station's movement is limited as result of the disturbing effect of solar gravitation which produces a systematic reduction in the altitude of orbital perigee. Consequently, having completed a certain number of turns, the station during alternate return to Earth will enter the dense layers of the atmosphere and cancel out its existence.

The magnitude of altitude reduction of the perigee per one rotation depends first of all upon the altitude of the apogee and can rise sharply during increase of same. When selecting the trajectory of an interplanetary station it was necessary to tend that the altitude of the apogee should be possible lower and not by much exceed the distance from Earth to the Moon. It was also necessary to secure sufficiently greater altitude of the perigee during the first round trip of the station around the Earth. Upon the degree of fulfilling both these requirements depends the total number of turns of the automatic station around the Earth and the time of its existence.

The influence of the Moon is not limited by this effect, which it produces in the period of first approach. Disturbances in the orbit of the station due to lunar attraction are not of such regular nature as are the disturbances due to solar attraction, and depend to a large extent upon the period of rotation of the station around the Earth. The influence of the Moon can become substantial if during any of the following turns the trajectory of the automatic station will again pass quite close to the Moon. The nature of the station's movement can change here considerably. If the interplanetary station will pass around the Moon from southern direction, i.e. the approach will be of the very same type as the first one, then there will be a sharp increase in the number of turns and in the life span of the station, by

preserving the basic property of its trajectory- approaching the Earth from direction of northern hemisphere. If the pass will be made from the northern side then the altitude of the orbital perigee decreases, and in the case of a sufficiently strong disturbance, the station may enter the terrestrial atmosphere during its next return to it.

On these orbital loops, where there is no sufficiently close approach to the Moon, the Moon nevertheless exerts a certain influence on the movement of the station. In spite of the fact that lunar attraction is very small in this case, but acting against a considerable number of trajectory loops, the attraction of the Moon exerts a noticeable influence on the movement of the automatic station, causing a reduction in the altitude of the perigee and cutting the time of station's existence in the orbit.

The chart showing the movement of the automatic interplanetary station under the influence of simultaneously acting gravitational forces of the Earth, Moon and Sun is a highly complex one. The determinant facter for the entire movement of the interplanetary station is the nature of its passing near the Moon during first approach.

Since absolutely no correction is made in the movement of the interplanetary station during the trip and the entire flight is determined finally by the parameters of motion at the end of the separtion section, then the realization of the above described trajectory of the station is possible only with an extremely accurate control system of the carrier rocket.

It can be imagined that through the center of the Moon perpendicular with Earth-Moon line runs a plane, which we will call the chart plane. The features of trajectory passing relative to the Moon can be characterized by the position of the point of intersection between trajectory and pictorial plane.

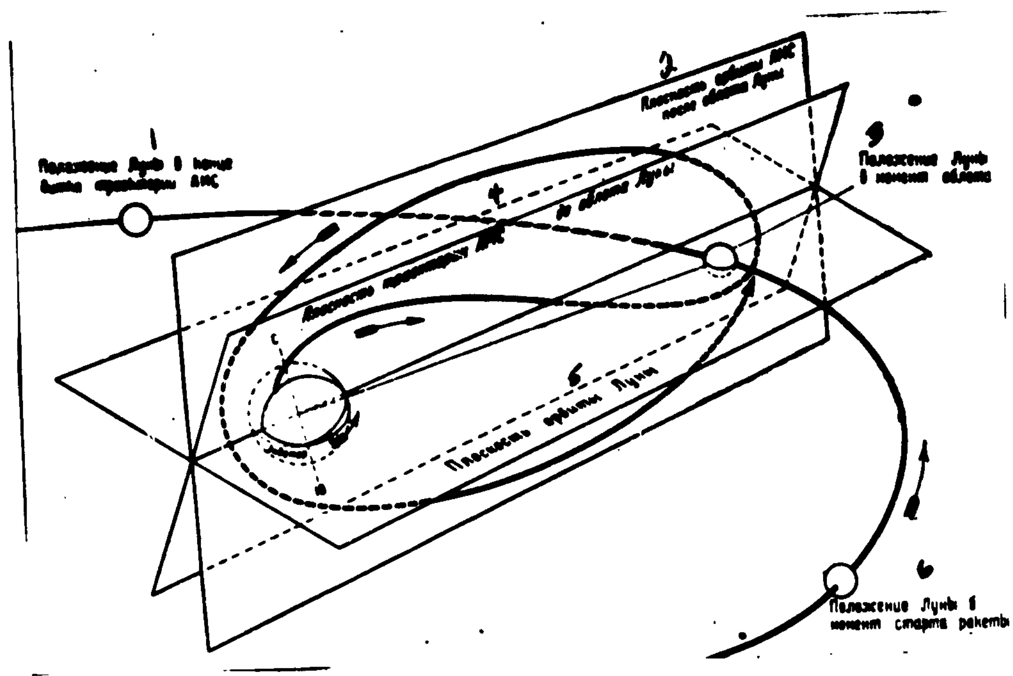

Fig. 108. Schematic drawing of flight trajectory of AIS.

1- position of Moon at the end of AIS trajectory loop; 2- plane of AIS orbit after rounding the Moon. 3- Plane of AIS trajectory prior to rounding Moon; 4- Position of Moon at the moment of rounding; 5- plane of lunar orbit; 6- position of Moon at the moment of rocket starting.

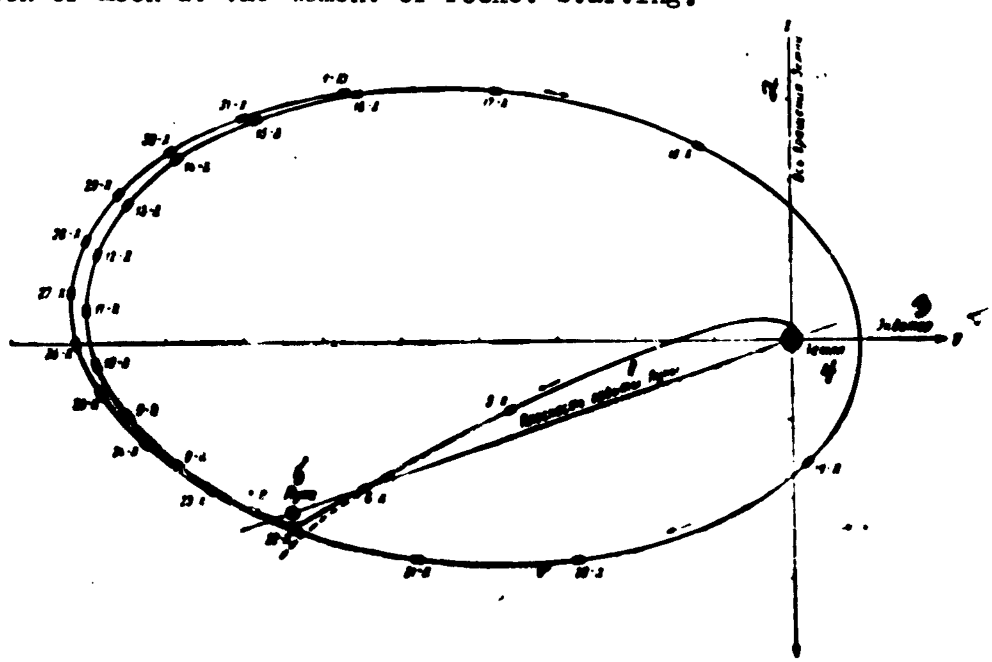

Fig. 109. Flight trajectory of AIS. View from the side of Spring equinox point. 1-plane of lunar orbit; 2- axis of Earth's rotat~~ion; 3-~~ Equator; 4- Earth; 5- Moon.

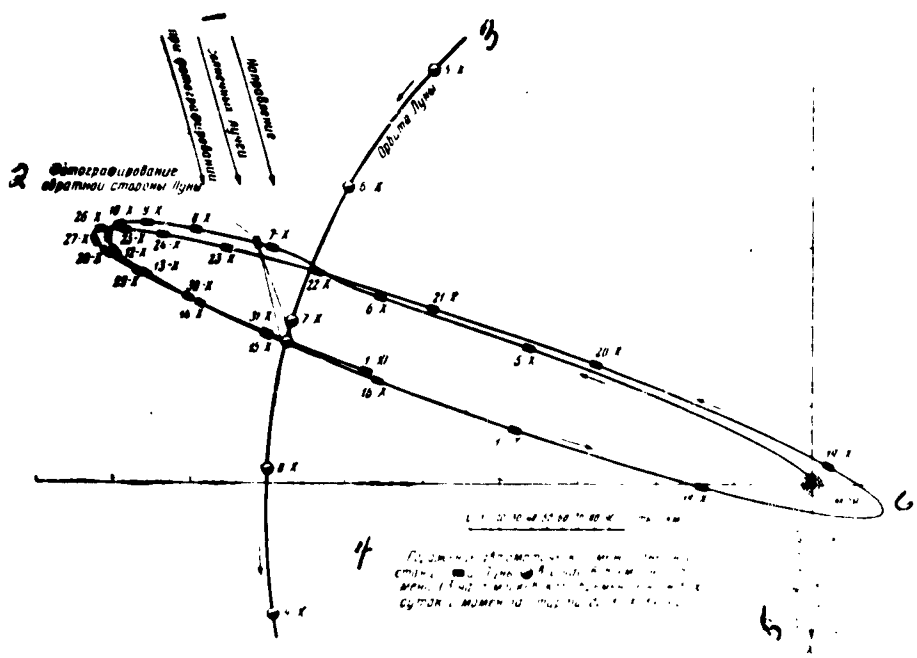

Fig. 110. Flight trajectory of AIS. Projection on the plane of terrestrial equator

1– Direction of solar rays during the photographing; 2– photographing reverse side of Moon; 3– orbit of Moon; 4– position of AIS and Moon o at zero hours universal time (3 o'clock Moscow time) during each diurnal period from starting moment to 11-1-1959; 5– direction toward point of spring equinox; 6– Earth.

Calculations show that during deviation of the intersecting point between trajectory and pictorial plane from the nominal (rated ) position by 1000 km the minimum distance between Earth and station at the end of the first round may change by 5 - 10 thousand km, and the time of returning to Earth- by 10-14 hours.

Requirements for the accuracy at terminal point remain so rigid as in the case of striking. This is basically connected with the fact that errors in the velocity magnitude toward the end of termination point in the case of an elliptical round flight trajectory cause deviations in the intersecting point between trajectory and pictorial plane which are 3-4 times greater than in the case of a hyperbolic trajectory,which are advisable to use during striking.

The disturbing effect of the Moon when passing close to it intensifies the influence of deviations of movement parameters at the end of the orbiting section on the nature of station's movement during its return trip to Earth after the flight around the Moon. That is why even the slightest errors in determining these parameters lead to very substantial errors in calculating the characteristics of motion of the AIS during its return to Earth.

The trajectory of motion of the AIS is shown in fig.109-111.

On October 5, 1959 at 20 hrs Moscow time the station was away at a distance of 284 thousand km from the Earth. At 16 hrs,16 min on October 6 it was at the shortest distance from the Moon,equalling 7900 km.

The AIS coordinates during its further movement are given in table 29.

Data concerning the position of ship (AIS) during photographing,established as result of processing trajectory measurements,data necessary for tying down the detected objects on the invisible side of the Moon,to the selenographic to the coordinate grid, are listed in table 30.

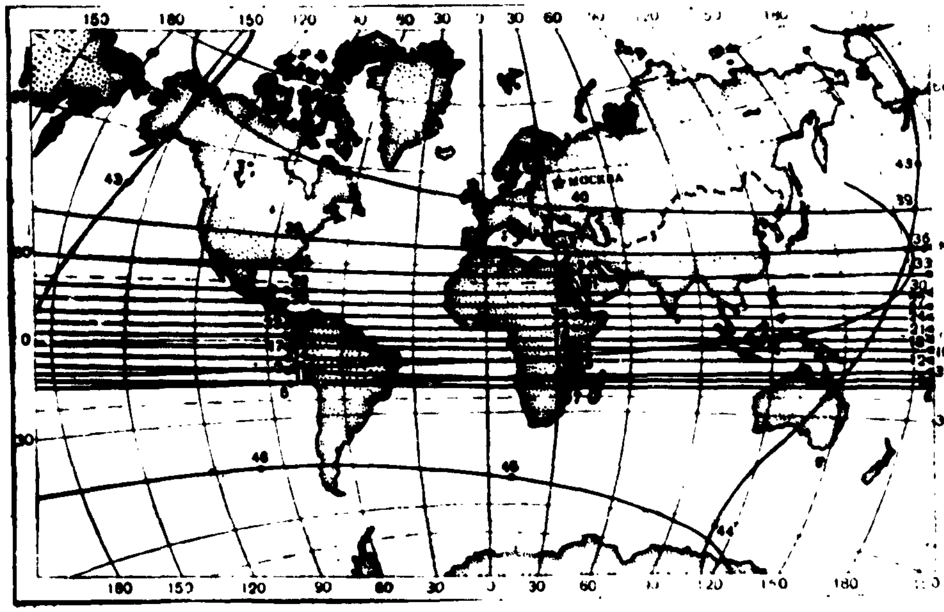

Fig.111.Projection of orbit of AIS on Earth

Table 29.Coordinates of AIS during its movement after coming close to the Moon

| Date | Distance from Earth in thousands of km | Inclination | Direct ascent |
|---|---|---|---|
| 7/X-59 g.20ch. | 417 | -11°36' | 16h32' |
| 8/X-59 g.20ch. | 448 | - 6°48' | 16h36' |
| 9/X-59 g.20ch. | 466 | - 2°36' | 16h40' |
| 10/X-59 g.20ch. | 470 | 1°23' | 16h44' |
| 13/X-59 g.20ch. | 430 | 13°54' | 16h55' |
| 16/X-59 g.20ch | 267 | 34°53' | 16h15' |
| 18/X-59 g.20ch. | 40 | | |

Table 30.Coordinates of AIS when photographing the Moon

| | Date | Moscow time | Distance from center of Moon km | Selenographic projection latit | longit |
|---|---|---|---|---|---|
| Beginning of photo graphing | 7/X-1959 | 6h30' | 65 200 | 16,9° | 117,6° |
| Completed photo graphing | 7/X-1959 | 7h10' | 68 400 | 17,3° | 117,1° |

Processing of trajectory measurements allowed to establish that the AIS complet ed in orbit about 11 trips around the Earth.



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Determining the Density of the Upper Atmosphere by Means of the Diffusion of Sodium Vapor.

A new original method for determining atmospheric density at great heights is the artificial-comet method developed by Soviet scientists. This method utilizes the phenomenon of the artificial radiation by certain gases of individual spectral lines and bands (characteristic of the given gas). This phenomenon is called resonance fluorescence. As caluclations have shown, sodium is the most favorable element for creating an artificial comet.

A sodium cloud which scatters the sun's rays is an exceptionally powerful source of light. A 1 kg mass of sodium vapors has a power of about 700 kilowatts.

An invaluable advantage of a sodium cloud is that it scatters light of a strictly defined wavelength $\lambda = 0.589\ \mu$ (the yellowish-orange region of the spectrum). This makes it possible, using suitable light filters, to observe the sodium cloud even when projected against a quite bright sky.

This method was verified experimentally by a geophysical rocket sent aloft at 430 km, and by the first and second space rockets. Since it is an absolute optical method for observing the movement of objects in space, the sodium comet makes it possible to determine the thickness of the medium in which the artificial comet forms (Figure 116). Thus, the density of the atmosphere at 430 km was determined on the basis of the diffusion theory.

The average displacement of a particle due to diffusion is proportional to the mean free path ($\bar{l}$) and the number of collisions (n) in time t:

$$s = \sqrt{\frac{n}{3}} \cdot \bar{l}, \qquad (4.8)$$

Noting that $n = vt/\bar{l}$ (where v is the mean particle velovity), we have:

$$s = \sqrt{\frac{1}{3} v \bar{l} t}. \qquad (4.9)$$

The value $\frac{1}{3} v \bar{l}$ is determined directly from observation data, and $0.85 \cdot 10^{11}$

cm$^2$/sec. For sodium atoms at a termperature of 1600° we have

$$v = 1.5 \cdot 10^5 \text{ cm/sec.}$$

Thus we get $\bar{l} = 1.7 \cdot 10^6$. But, on the other hand,

$$\bar{l} = \frac{1}{n_1 \cdot \bar{Q}_d}, \qquad (4.10)$$

where $n_1$ is the concentration of atmospheric atoms; and $\bar{Q}_d$ is the effective cross section of diffusion. Taking $\bar{Q}_d = 3.85 \cdot 10^{-15}$ g/cm$^3$, with an accuracy to 20–30%, we get

$$n_1 = 1{,}6 \cdot 10^8 \text{ см}^{-3}. \qquad (4.11)$$

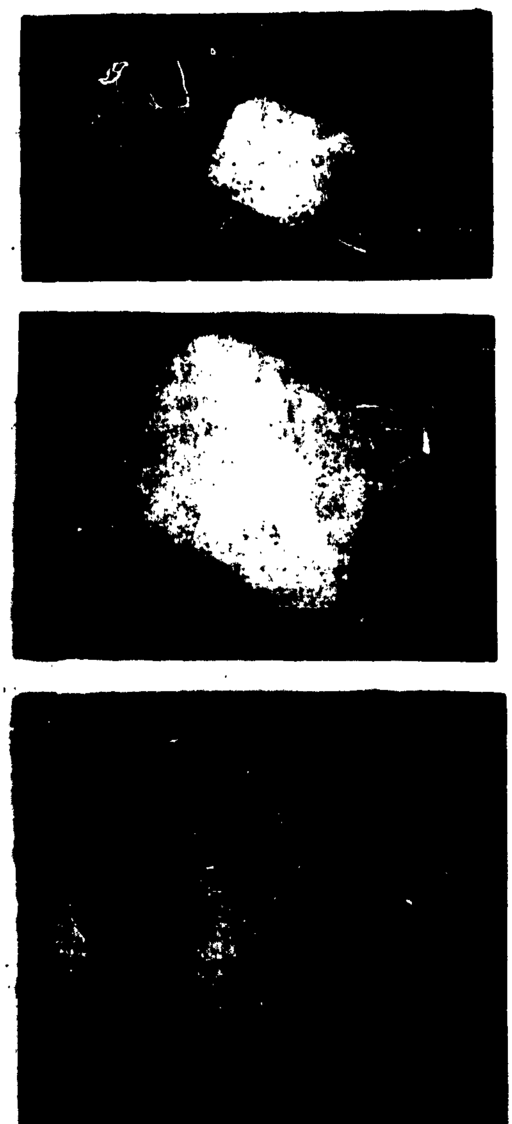

Figure 116. Various Stages in the Formation of a Sodium Cloud at 430 Kilometers.



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Considering that the composition of the atmosphere to 500 km is basically nitrogen and oxygen, we get for the density at 430 km,

$$\rho = 4{,}7 \cdot 10^{-15}\ g/cm^3. \qquad (4.12)$$

A more accurate quantitative calculation, based on solution of the differential diffusion equation, gives the following values:

$$n_1 = 2{,}5 \cdot 10^8\ cm^{-3},$$
$$\rho = 6{,}7 \cdot 10^{-15}\ g/cm^3. \qquad (4.13)$$

The errors in these measurements are no more than 30%.

## Manometer Measurements of Density and Pressure

In addition to determining the characteristics of the upper atmosphere by study of the deceleration of satellites, the diffusion of sodium vapor (artificial comet), and observations of the radio signals from satellites, the third Soviet artificial satellite was equipped to take direct measurements of pressure and density at various heights using ionization and magnetic electric-discharge manometers. The experimental method was the same as described in Chapter II. Figures 117 and 118 show the apparatus aboard the third satellite for measuring pressure.

The magnetic electric-discharge manometer (Figure 117) was designed to measure pressure in the range $10^{-5}$ to $10^{-7}$ mm Hg, and the ionization manometers (Figure 118) measured pressure from $10^{-7}$ to $10^{-9}$ mm Hg.

Figure 117. Magnetic Manometer.

Figure 118. Ionization Manometers.



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To avoid the influence of ionospheric ions and electrons on the measurement results, the ionization manometers were equipped with special shields and traps. All manometers have avacuated and sealed flasks which are opened by special mechanisms after the satellite has been ejected into orbit. During the entire pressure-measurement time the amplifiers were periodically calibrated, the emission current measured, and the wall temperature of one of them was measured.

A number of theoretical and laboratory investigations were conducted during the preparation of the experiment. Much attention was devoted to determining the gas release of the satellite; this was done by measuring the duration of the degasification of the structural materials, and also by developing methods to insure maximum hermetic sealing of the satellite.

Because of the duration of the measurements made aboard the satellite, it was possible by means of the manometer readings to study the degasification of the outer surface of the satellite and to fix the time at which the gas release of the satellite begins to influence the measurement results.

In determining the pressure from the readings of the manometers aboard the satellite, it was necessary to take into account its orientation in space, its velocity, and the gas composition and temperature.

For final interpretation of the manometer readings it was necessary to establish a relationship between the pressure inside the manometer, measured and transmitted to earth, and the pressure of the external medium. Such relation can be obtained theoretically by using the laws of molecular aerodynamics (See Appendix 4). The pressure $P_1$ measured by the manometer is associated with the particle concentration N at a given point in the atmosphere by the ratio

$$N = \frac{A \cdot P_1}{\sqrt{2\pi M R T_1} V \sin\theta}, \qquad (4.14)$$

where A is Avogadro's number; R is the gas constant; M is the molecular weight; V is the velocity of the satellite; θ is the angle between the velocity vector

of the satellite and the plane of the manometer aperture; $T_1$ is the temperature of the manometer wall.

After determining N, we calculated the density and height of a uniform atmosphere:

$$H = \frac{\Delta h}{2{,}3(\lg N'' - \lg N')}, \tag{4.15}$$

where N" and N' is the concentration of particles in a unit volume of the atmosphere at two points, one 10 km above the other.

We then calculated temperature T:

$$T = \frac{MgH}{R} \tag{4.16}$$

and external pressure P:

$$P = kNT \tag{4.17}$$

We used Formula (15) in Appendix 4 to calculate the orientation angle θ.

Analysis of the obtained data, with certain assumptions regarding the mean molecular weight M, made it possible to construct a cross section of the atmosphere for 225–500 km; the structural parameters of this atmosphere are given in Table 32.

The measurements were made for various times on May 16, 1958 (1300–1900 hrs local time) and for various geographic latitudes (57°N – 65°N).

The data in Table 32 are in good agreement with the density values obtained from the deceleration of satellites, the diffusion of sodium vapor, and the results of radio observations of satellites.

Figure 119 gives the results of density measurements using all the indicated methods.

Information on the density of the upper atmosphere ($\rho = 3 \cdot 10^{-13}$ g.cm$^{-3}$ at 225 km), obtained from the deceleration of the first artificial earth satellite and from the diffusion method ($\rho = 6.7 \cdot 10^{-15}$ g.cm$^{-3}$ at 430 km), has caused an essential change in our concepts as to the parameters of the upper atmosphere. These measurements show that the density of the atmosphere at 220 km

Table 32

**Structural Parameters of the Atmosphere at 225-500 km Altitude**

| Height, km. | $N$, $cm^{-3}$ | $\rho$, $g/cm^3$ | $H$, km | $T$, °K | $P$, $dyne/cm^2$ | $P$, mm Hg |
|---|---|---|---|---|---|---|
| 225 | $6,01 \cdot 10^9$ | $2,12 \cdot 10^{-13}$ | 40,0 | 936 | $7,76 \cdot 10^{-4}$ | $6,25 \cdot 10^{-7}$ |
| 230 | 5,31 | 1,79 | 40,6 | 938 | 6,88 | 5,54 |
| 235 | 4,7 | 1,7 | 41,3 | 941 | 6,1 | 4,92 |
| 240 | 4,17 | 1,42 | 42,0 | 946 | 5,44 | 4,4 |
| 245 | 3,71 | 1,25 | 42,8 | 952 | 4,88 | 3,94 |
| 250 | 3,3 | 1,1 | 43,5 | 958 | 4,36 | 3,54 |
| 255 | 2,94 | $9,73 \cdot 10^{-14}$ | 44,3 | 964 | 3,91 | 3,17 |
| 260 | 2,64 | 8,66 | 45,2 | 971 | 3,54 | 2,88 |
| 265 | 2,36 | 7,77 | 46,0 | 979 | 3,19 | 2,6 |
| 270 | 2,12 | 6,83 | 47,0 | 987 | 2,89 | 2,35 |
| 275 | 1,91 | 6,1 | 47,9 | 996 | 2,63 | 2,14 |
| 280 | 1,72 | 5,44 | 48,8 | 1005 | 2,39 | 1,95 |
| 285 | 1,55 | 4,87 | 49,7 | 1015 | 2,17 | 1,78 |
| 290 | 1,4 | 4,36 | 50,7 | 1026 | 1,98 | 1,62 |
| 295 | 1,27 | 3,93 | 51,7 | 1037 | 1,82 | 1,49 |
| 300 | 1,15 | 3,53 | 52,7 | 1048 | 1,66 | 1,37 |
| 305 | 1,07 | 3,26 | 53,7 | 1059 | 1,56 | 1,29 |
| 310 | $9,57 \cdot 10^8$ | 2,9 | 54,5 | 1072 | 1,42 | 1,17 |
| 315 | 8,73 | 2,63 | 55,9 | 1084 | 1,31 | 1,008 |
| 320 | 7,98 | 2,39 | 57,0 | 1097 | 1,21 | 1,0 |
| 325 | 7,31 | 2,17 | 58,1 | 1110 | 1,12 | $9,28 \cdot 10^{-8}$ |
| 330 | 6,7 | 1,98 | 59,2 | 1124 | 1,04 | 8,62 |
| 335 | 6,17 | 1,82 | 60,3 | 1138 | $9,69 \cdot 10^{-5}$ | 8,06 |
| 340 | 5,68 | 1,66 | 61,5 | 1153 | 9,04 | 7,52 |

Table 32 (cont.)

| Height, km | $N$, $cm^{-3}$ | $\rho$, $g/cm^3$ | $H$, km | $T$, °K | $P$, $dyne/cm^2$ | $P$, mm Hg |
|---|---|---|---|---|---|---|
| 345 | 5,22 | 1,52 | 62,8 | 1169 | 8,96 | 7,46 |
| 350 | 4,82 | 1,4 | 64,8 | 1185 | 7,88 | 6,58 |
| 355 | 4,46 | 1,29 | 65,2 | 1200 | 7,39 | 6,18 |
| 360 | 4,13 | 1,19 | 66,7 | 1219 | 6,95 | 5,82 |
| 365 | 3,86 | 1,1 | 68,1 | 1237 | 6,59 | 5,53 |
| 370 | 3,56 | 1,02 | 69,5 | 1257 | 6,18 | 5,19 |
| 375 | 3,31 | $9{,}41 \cdot 10^{-15}$ | 70,9 | 1276 | 5,83 | 4,9 |
| 380 | 3,08 | 8,72 | 72,4 | 1295 | 5,51 | 4,64 |
| 385 | 2,92 | 8,24 | 73,9 | 1305 | 5,26 | 4,44 |
| 390 | 2,69 | 7,56 | 75,2 | 1335 | 4,96 | 4,19 |
| 395 | 2,52 | 7,07 | 76,7 | 1353 | 4,71 | 3,98 |
| 400 | 2,36 | 6,6 | 78,9 | 1373 | 4,47 | 3,79 |
| 405 | 2,21 | 6,16 | 79,7 | 1393 | 4,25 | 3,6 |
| 410 | 2,08 | 5,78 | 81,2 | 1417 | 4,07 | 3,46 |
| 415 | 1,95 | 5,41 | 82,9 | 1440 | 3,83 | 3,3 |
| 420 | 1,84 | 5,09 | 84,6 | 1465 | 3,72 | 3,17 |
| 425 | 1,73 | 4,79 | 86,3 | 1489 | 3,56 | 3,04 |
| 430 | 1,64 | 4,51 | 88,1 | 1514 | 3,43 | 2,93 |
| 435 | 1,55 | 4,25 | 90,0 | 1539 | 3,29 | 2,82 |
| 440 | 1,47 | 4,03 | 91,7 | 1563 | 3,17 | 2,72 |
| 445 | 1,39 | 3,8 | 93,6 | 1589 | 3,05 | 2,62 |
| 450 | 1,32 | 3,6 | 95,5 | 1614 | 2,94 | 2,53 |
| 455 | 1,25 | 3,4 | 98,6 | 1643 | 2,84 | 2,44 |
| 460 | 1,19 | 3,23 | 99,9 | 1675 | 2,75 | 2,37 |
| 465 | 1,13 | 3,06 | 102,0 | 1709 | 2,66 | 2,3 |
| 470 | 1,08 | 2,92 | 104,5 | 1745 | 2,6 | 2,25 |
| 475 | 1,03 | 2,79 | 107,0 | 1781 | 2,53 | 2,19 |
| 480 | $9{,}82 \cdot 10^{7}$ | 2,65 | 109,3 | 1810 | 2,45 | 2,13 |
| 485 | 9,4 | 2,53 | 111,5 | 1845 | 2,39 | 2,08 |
| 490 | 8,97 | 2,42 | 113,9 | 1880 | 2,33 | 2,02 |
| 495 | 8,61 | 2,31 | 116,5 | 1917 | 2,28 | 1,98 |
| 500 | 8,24 | 2,21 | 119,0 | 1953 | 2,22 | 1,94 |

is greater during the day than at night, and greater in the polar regions than in the equatorial regions. Data obtained from the first artificial earth satellites allow us to draw conclusions only regarding the diurnal regime, as yet. The observed great deceleration of satellites can be explained by the fact that the temperature of the upper atmosphere is higher than that determined in older "rocket models".

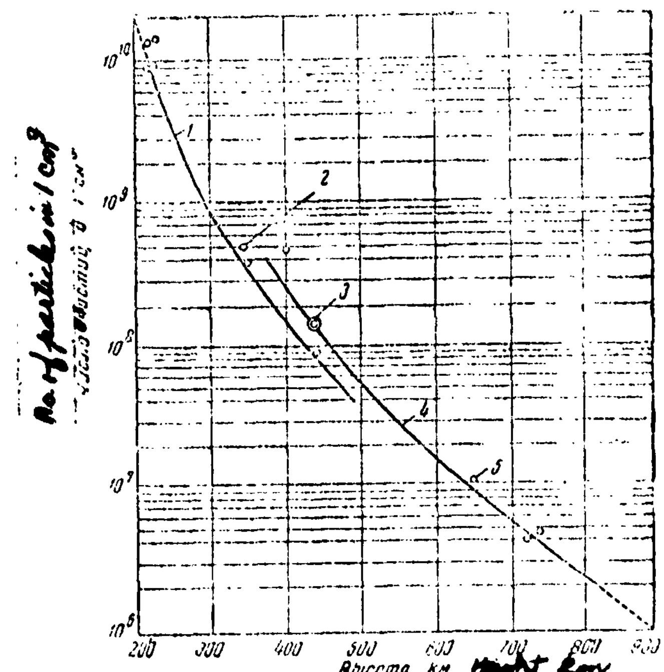

Figure 119. Results of Density Determination Using Various Methods: 1) third satellite (manometers); 2) satellite deceleration (Soviet studies); 3) sodium cloud (high altitude rocket); 4) first satellite (radio observations); 5) satellite deceleration (American studies).

This viewpoint is supported by extensive manometer measurements which make it possible to reliably determine the height of the uniform atmosphere at various levels in the upper atmosphere. Measurements showed that the height of the uniform atmosphere increases as the distance from the earth; this indicates a decrease in molecular weight M due to molecular dissociation, and a gradual increase in the temperature of the atmosphere. At 225-500 km the temperature was 1200-2000°K.

## Mass-Spectrometer Measurements of the Ion Composition of the Upper Atmosphere.

To measure the ion composition of the ionosphere the third Soviet satellite contained a radio-frequency mass-spectrometer, the principle of which has been covered in Chapter II. The mass-spectrometer was designed to record ions with mass numbers of 6 to 50. Figure 120 shows such an apparatus.

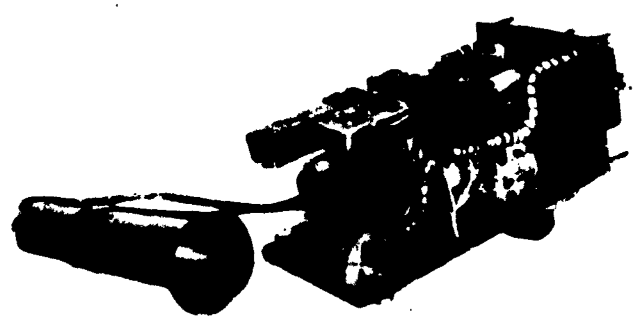

Figure 120. Radio-Frequency Mass-Spectrometer.

In the period 15-25 May 1958 approximately 15,000 mass spectra were obtained at heights ranging from 225 to 980 km. The measurements were made in the Northern Hemisphere in latitudes 27-65°. The data obtained were for the daytime hours (0700-1100 hrs Moscow time). The mass number M of the peak was determined from the formula

$$M = \frac{1}{k}\left(\frac{V^* + \varphi}{1 - 300 m_0 V^2/2qk}\right), \qquad (4.18)$$

where V* is the value of the scan voltage at the moment of the peak; $\varphi$ is the negative potential of the satellite; $\underline{k}$ is the instrument constant; $m_0$ is the mass of a hydrogen atom; $\underline{q}$ is the ion charge; and V is the satellite velocity.

The main difficulty in decoding the data is in separating the basic (true) ion masses from the harmonic (false) masses. The data showd that at 225-980 km the most intense ion is the peak with mass number 16, i. e., an ion of atomic oxygen $O^+$. The second most intense ion is that with a mass number of 14, i. e.,

an ion of atomic nitrogen $N^+$.

In the region of the perigee there is a group of peaks with mass numbers 32, 30 and 28. Most of these are peaks with mass number 30, i. e., a nitric oxide ion $NO^+$. Peaks with mass numbers 32 and 28 are ions of molecular oxygen $O_2^+$ and molecular nitrogen $N_2^+$. If we compare the intensity of all mass ions with that of an ion of atomic oxygen, the picture is as follows:

The ratio $J_{N+}/J_{O^+}$ at 230-650 km varies from 1.3 to 8-10%, depending on height and geographic latitude (Figures 121 and 122); the ratio of the intensities $J_{O_2^+}/J_{O^+}$ in the southern latitudes varies from 2.5 to 8% at 250-230 km (to the perigee), while in the northern latitudes the peak of ions of molecular oxygen can be traced to 400 km; the ratio at these heights $J_{O_2^+}/J_{O^+} \sim 0.1\%$; at 400-500 km the ratios $J_{NO^+}$ and $J_{N_2^+}/J_{O^+}$ are practically identical, 0.2-1%. Above km no molecular ions can be found, and the ionosphere becomes purely atomic, with an accuracy to 0.1%, an oxygen-nitrogen atmosphere.

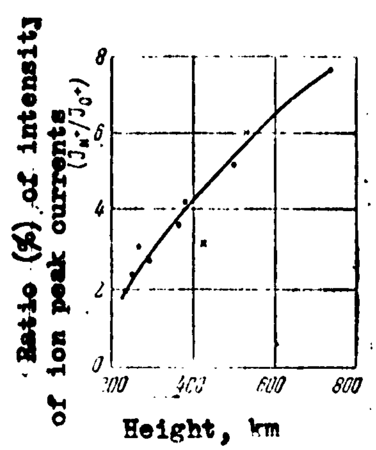


Figure 121. Change of the Relative Intensity of the Ion Peaks of Atomic Nitrogen as a Function of Height, from the Data of Two Orbits on 23 May 1958.

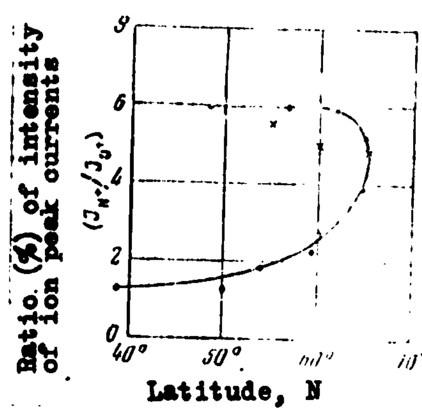


Figure 122. Change of the Relative Intensity of the Ion Peaks of Atomic Nitrogen vs. Geographic Latitude, from the Data of Two Orbits on 23 May 1958.

Up to 250 km, the most wide-spread ions are $NO^+$ ions which form, according to available information, as a result of the reaction of oxygen ions with

neutral molecules of nitrogen, or as the result of the reaction of oxygen atoms with ionized nitrogen molecules. Ion composition measurements made aboard the third satellite showed that at heights to 500 km there are molecular ions, while above 500 km there are only atomic ions. This result is important from the viewpoint of explaining the processes of ionization balance in the atmosphere. Another interesting fact is the noticeable concentration of $O^+$ ions at heights of the order of 1000 km. Hydrogen is not the basic (predominant) component of the ionosphere right up to 1000 km, which changes our previous concepts.

## INVESTIGATION OF THE IONOSPHERE

A study of the propagation of radio waves in the ionosphere, the extent of their absorption, and determination of ionisation of the upper atmosphere.

Extensive materials with recordings of radio signals from artificial earth satellites have been accumulated. These observations were carried out at points located at different geographic latitudes and longitudes, by radio-direction stations, by DOSAAF clubs, by a number of institutions of higher learning, and by thousands of radio amateurs. It is known that owing to the electromagnetic properties of the ionosphere, radio waves are propagated over very great distances. In this connection we can point out one interesting phenomenon which was known earlier, but which was especially manifested clearly during observation of the signals of the Soviet artificial earth satellites. This phenomenon is called the antipodal effect and consists in the following: the power of the received signal increases at a point located at the antipode of the transmitting station. From the recordings of the results of the reception of radio signals of the first satellite in the Antarctic, in the village of Mirnyy, we see (Figure 123) how the radio signals of the satellite were received on a frequency of 20 Mc when it was in the region of Mirnyy and at the antipode to it. Such cases, when over a long period of time favorable conditions are realized in the ionosphere for "run-off" of radio waves to the diametrically opposite point on earth, are of considerable interest.

The measurements of the strength of the field of radio signals received from the satellite are of very great significance. The results of measuring the field strength of the radio signals permits us to estimate the absorption of radio waves in the ionosphere, including those regions which lie above the main maximum of ionization of the ionosphere of the F-2 layer, and therefore are inaccessible to ordinary measurements conducted on the earth's surface.

These measurements also permit us to judge the possible pathways for the propagation of radio waves in the ionosphere. The results of receiving the satellite's signals and the measurements of their levels show that these signals on the 15-m wave were received over very great distances, far exceeding the distances of straight visibility. These distances reached 10, 12 and 15 and sometimes more thousands of kilometers.

Of especial interest is that a satellite completing motion about an elliptic orbit, occupies a different position relative to the main $F_2$ maximum. When processing the material on radio observations we must take into consideration whether the satellite at a given instant of time is above or below the true height of the $F_2$ maximum which is obtained on the basis of the high-frequency characteristics of the ionosphere recorded by the ionospheric stations. If in the Southern Hemisphere the satellite moves above the $F_2$ layer, then in the Northern Hemisphere it is sometime above the ionisation maximum of this layer and at other times below it, and again close to this maximum. Such conditions create a great diversity in the pathways of propagating short radio waves over considerable distances.

We have already spoken above about the antipodal effect in the propagation of radio waves. Another of the possible ways to propagate radio waves is their reflection from the earth's surface having passed through the entire thickness of the ionosphere with subsequent single reflection from the ionosphere in those regions where the critical frequencies are sufficiently great. In other cases the radio waves striking at a certain angle from above upon the ionosphere undergo an appreciable refraction and thus penetrate the region lying beyond the limits of geometric straight visibility.

A satellite position close to the region of maximum atmospheric ionisation creates especially favorable conditions for propagation of radio waves by ionospheric wave guides. In certain cases the radio waves arrived at the receiving point, not over extreme distances, but by short-cutting the earth over

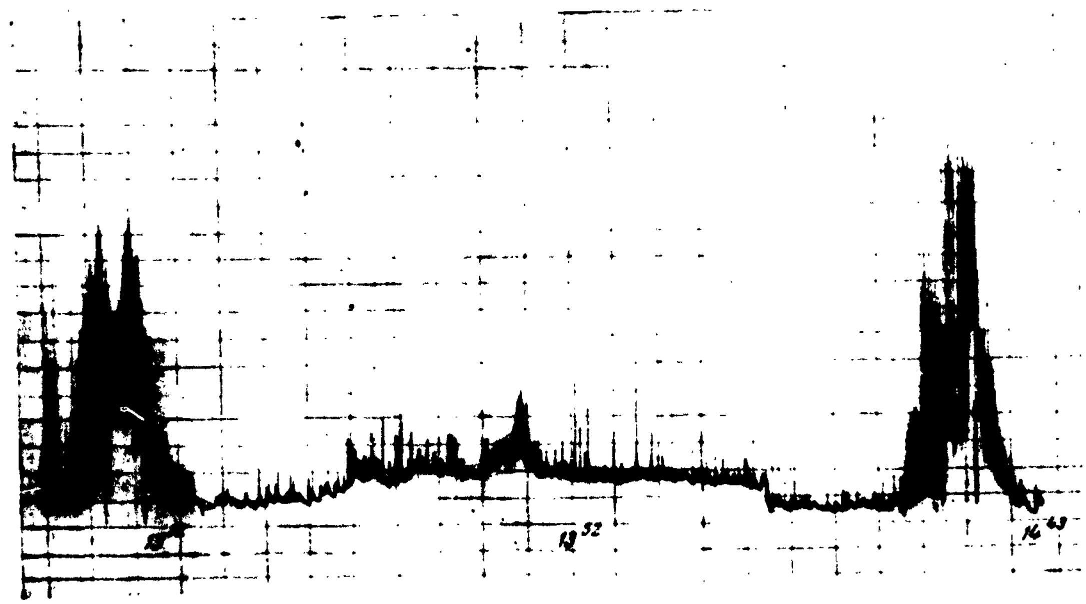


Figure 123. Recording of Radio Signals when Satellite is at Antipode.



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a longer arc of the great circle. In individual cases we observed the phenomenon of around-the-world echo of radio waves. In some cases the measured values of the field strength proved to be larger than those calculated by the law of inverse proportionality of the first power of distance, which also proclaims the presence of channel wave guides in the ionosphere.

The described phenomena — the distortion of the pathways of radio-wave propagation, their reflection, partial or complete absorption — are determined by the conditions of the ionosphere and, in particular, by the value of the electron concentration which is one of the main parameters of the ionosphere.

Until recently the electron concentration was measured mainly within heights up to 300 km, i. e., below the principal maximum of the $F_2$ layer. The greatest value of the electron concentration, detected in the middle latitudes, reached 2-3 million electrons per 1 $cm^3$. In addition, the electron concentration increases with height: at 300 km it is 10-15 times greater than at a height of the order of 100 km.

With the development of artificial satellites, new possibilities were manifested for an efficient study of the ionospheric layers lying above the ionisation maximum. The observation method of "radio rising" and "radio setting" of the satellite was used to study the distribution of the electron concentration with height. This method of radio observation consists of the following.

As the satellite moves around the orbit, the trajectory of the signals being received in the ionosphere between the satellite and the observation point has the form depicted in Figure 124 (curves 1,2,3). We assume that the satellite emitting radio waves of frequency $\omega$ pass over the observation point above the principal maximum of the electron concentration of the ionosphere ($N_M$), the critical frequency of which $f_c$ is determined from the relation:

$$\omega_c^2 = (2\pi f_c)^2 = \frac{4\pi e^2}{m} N_M = 3,18 \cdot 10^9 N_M. \qquad (4.19)$$

If $\omega > \omega_c$, then the propagation of radio waves close to the optical and the

corresponding trajectories are straight lines 1', 2', 3'; as is well known, the visible optic "setting" or "rising" is characterised by the fact that a light ray coming from the observed body is straight, tangent to the observation point.

1- satellite
2- optical setting
3- radio setting
4- max. horizontal range of receiving satellite signals

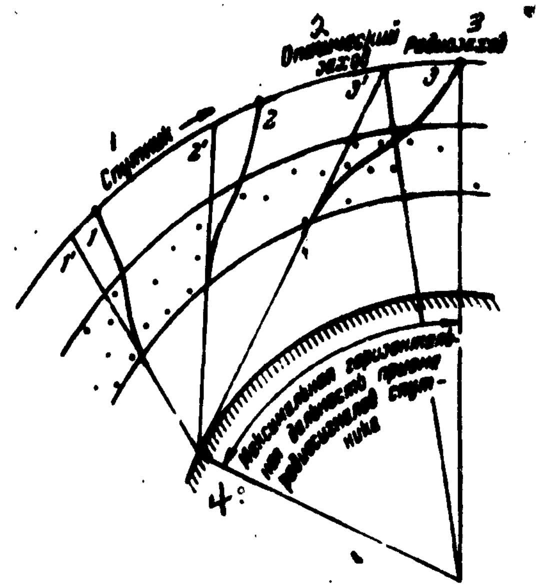


Figure 124. Trajectory of Radio Waves in the Ionosphere between the Satellite and the Observation Point.

If the value of $\omega$ is not very large, then owing to the curve of the trajectory of the wave in the ionosphere, the radio beam is not a straight line (curve 3). Therefore, the "radio setting" occurs later than the optical, and the "radio rising", conversely, is earlier than the optical. Knowing now the height of the satellite and the condition of the ionosphere up to its main maximum from the data of the ground ionosphere stations, we can calculate the electron concentration above the main maximum of the ionosphere. The method indicated was used to work out the results of the radio observations at six points during 5, 6, 7 October 1957. As a result the distribution of the electron concentration up to a height of 600-650 km was obtained. The electron

concentration in the outer ionosphere decreases with height considerably less slowly than it increases in its lower part. Its rate of change is slowed by about 5-6 and more times. Moreover, the data obtained permit us to calculate the value of the density of neutral particles (n). If we assume that for $h > 400$ km the life $\tau_e$ of a free electron reaches $10^5 - 10^6$ sec, then under quasi-stationary conditions, the ratio n/N has the value $\tau_n/\tau_e$ where $\tau_n$ is the time between the individual acts of ionisation. Hence

$$n \sim N\frac{\tau_n}{\tau_e}.$$

Table 33

**Value of Electron Concentration and Density of Neutral Particles Obtained from the Recordings of Radio Signals**

| Height Z, km | Electron conc. N, $cm^3$ | Density of neutral particles n in $cm^3$ |
|---|---|---|
| 200 | $10^5$ | — |
| 320 | $1,8 \cdot 10^6$ | — |
| 400 | $1,4 \cdot 10^6$ | $6 \cdot 10^8$ |
| 600 | $7 \cdot 10^5$ | $10^7$ |
| 1150 | $1 \cdot 10^5$ | $2 \cdot 10^5$ |
| 1800 | $1 \cdot 10^4$ | $2 \cdot 10^3$ |
| 2450 | $1 \cdot 10^3$ | 20 |
| 3100 | $1 \cdot 10^2$ | $< 1$ |

The results of the calculations are shown in Table 33 and in Figure 125 (the values of n and N above 650 km were obtained by extrapolation).

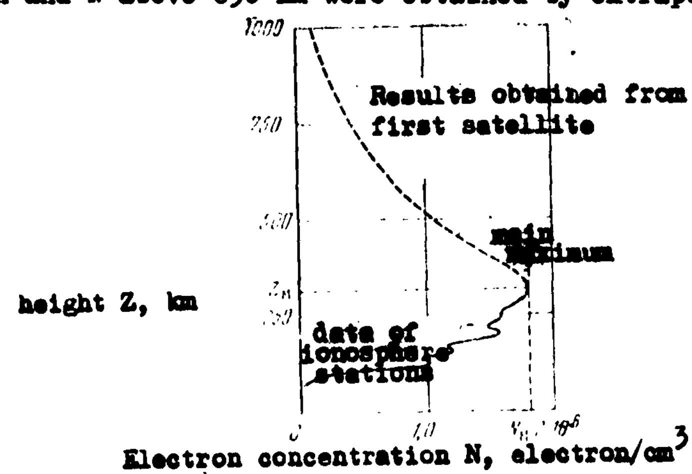


Figure 125. Curve of the Dependence of Electron Concentration of Ionosphere on Height above Earth's Surface.

## Determination of Concentration of Positive Ions in Upper Atmosphere.

Along with using radio methods, an experiment was set up on the third Soviet artificial satellite for the direct measurement of the concentration of positive ions up to heights of 900-1000 km. For this purpose a special device was installed on the satellite having two ion traps (Figure 126). The principle of the operation of such a device is cited in Chapter 2. Along with the concentration of positive ions, it permitted the determination of the potential of the satellite relative to the ambient medium. In the sections of the orbit illuminated by the sun, the potential proved to be minus 1-7 v. The value of the negative potential of the satellite can apparently be interpreted as the result of the effect on it of fast electrons whose energy considerably exceeds the average energy of the atmospheric particles.

Figure 126. Device for Measuring the Concentration of Positive Ions.

As a result of treating the material obtained, graphs were plotted of the variations in concentrations of positive ions based on the orbital loops.

The over-all view of the curves obtained by different methods has the same character to heights of 660, 760, 800 and 900 km over the earth's surface.

Of greatest interest is the curve which shows the variation of the ion concentration up to 1000 km (Figure 127).



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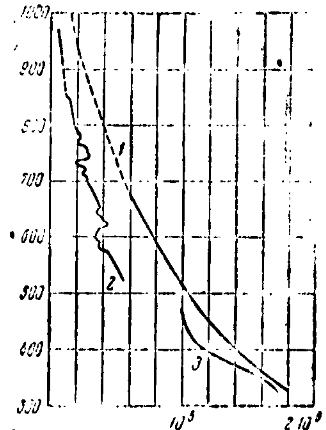

height, km

Number of electrons or positive ions per 1 $cm^3$.

Figure 127. Curves of the Variation of Electron Concentration Obtained by Different Methods: 1) first satellite, 5-8 Oct 1957, 7 hr 40 min - 9 hr 40 min (radio signals); 2) third satellite, 19 May 1958, 11 hr 00 min (ion trap); 3) high altitude rocket, 21 Feb 1958, 11 hr 40 min (radio inteferometer).

Starting at 850 km and higher we noted a considerable slowing down of the drop in the concentration of positive ions with height. The concentration of positive ions at 980 km is $6 \cdot 10^4$ ions/$cm^3$ provided that the ions are those of atomic oxygen. This assumption is experimentally confirmed by measurement of the ion composition at these heights.

We can assert with complete confidence that the concentration of about $3.6 \cdot 10^3$ ions/$cm^3$ is the lower limit of the concentration of positive ions at a height of about 1000 km.

<u>Detection of Electrons with an Energy of about 10 kev in the Upper Atmosphere.</u>

One of the factors of additional ionisation of the atmosphere are the corpuscular streams moving from the sun — fast protons, $\alpha$-particles, electrons, etc. These corpuscular streams penetrate the earth's atmosphere mainly in the polar regions at high geomagnetic latitudes, which is explained by the effect of the earth's magnetic field on them.

When the intense corpuscular streams penetrate the upper layers of the atmosphere there usually take place the phenomena of polar auroras, the observations of which have served until recently as the main method of investigating corpuscular radiation. The data obtained as a result of analyzing the spectra of polar auroras permitted us to make the assumption that in the upper atmosphere, owing to the variable magnetic fields generated by the interplanetary medium and by the solar corpuscular streams, there can be an acceleration of atmospheric electrons to an energy exceeding the energy of the electrons in the solar corpuscular streams. However, the constant presence of not especially hard corpuscles — electrons even over the low latitudes was not assumed and was associated only with corpuscular outbreaks in the zone of polar auroras.

An experiment was set up on the third Soviet satellite for the direct detection of not especially hard electrons in the upper atmosphere. For this purpose a device was used which recorded the corpuscles by means of a fluorescent screen and photomultiplier (Figure 128). The operating principle of this device was cited in Chapter II. The thin fluorescent screens of zinc sulfide activated by silver containing 2 mg/cm$^2$ of the substance made them insensitive to x-radiation generated by the electrons in the atmosphere and in the body of the satellite. To suppress protons with energies of several tens of kev, aluminum foil sheets, containing 0.4 and 0.8 mg of the substance per 1 cm$^2$, were placed in front of the screen.

By means of this device, not especially hard electrons with an energy of about 10 kev were detected directly for the first time. They were recorded at heights from 470 to 1880 km over sea level. Their intensity during the daytime was greater than during the night. In addition, the intensity continuously changes, considerably increased with height and over high geomagnetic latitudes. The least intensity was recorded over the geomagnetic equator. The electrons detected, as a rule, were moving close to directions perpendicular to the magnetic lines of force. The current densities developed by the electron streams

in directions perpendicular to the magnetic lines of force were in most cases an order greater than in a direction along and opposite to the magnetic lines of force. The current density towards the earth is apparently somewhat greater than in the opposite direction (Figures 129 and 130).

The energy flux of not especially hard electrons at the threshold of the device's sensitivity was about one-millionth of the flux of solar energy impinging per unit area of the earth's surface. When it exceeds the scale of the device it is equal to approximately one-thousandth of the energy flux of solar radiation.

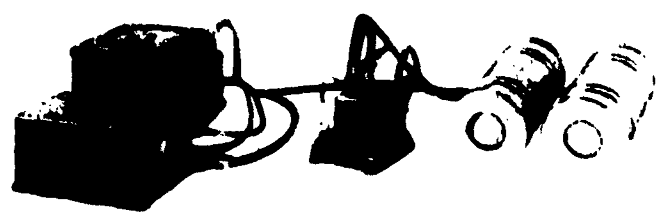

Figure 128. Device for Investigating Corpuscular Radiation.

The electrons, thus recorded, could not be directly by solar corpuscles since their velocity greatly exceeds the velocity of the motion of solar corpuscles determined from the observations of polar auroras. They most probably can be relegated to atmospheric electrons accelerated in the outer atmosphere due to the variable geomagnetic fields.

The new phenomenon detected is of great interest from the point of view of the physics of the upper atmosphere. It can explain a number of anomalies in the ionosphere and be an additional source of heating of the upper atmosphere over the polar regions.



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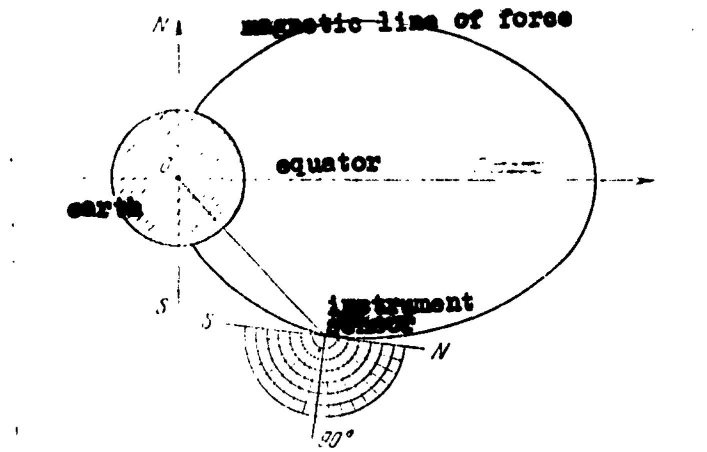


Figure 129. Diagram of the position of the instrument in space with an indication of the orientation of the sensors relative to the magnetic lines of force. The intensity of the recorder's irradiation by particles is shown in Figure 130 in polar coordinates related with the instrument. The polar angle characterizes the axial direction of the sensor with the magnetic lines of force.

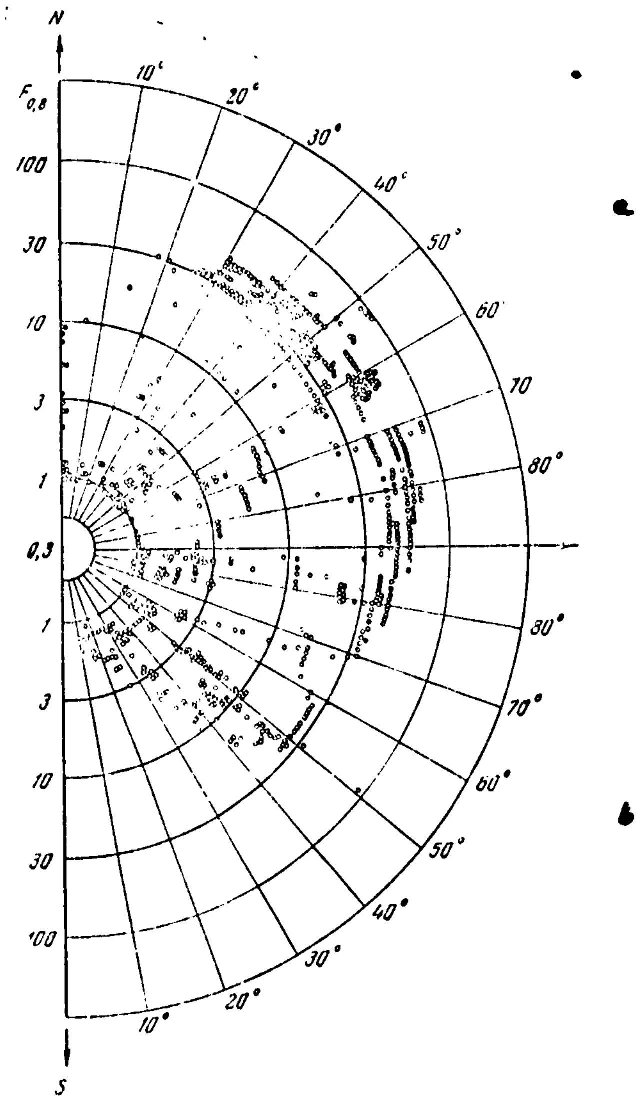


Figure 130. Irradiation intensity of the recorder with foil 0.8 mg/cm$^2$ by particles relative to the orientation of the sensor with respect to the magnetic lines of force (the intensity in conventional units is laid out along the radius).
a) sensor recording the corpuscular stream coming along the lines of force toward the earth; b) sensor recording the corpuscular stream coming along the lines of force from the earth.

A Study of Radiation Near the Earth and in Cosmic Space.

The second and third Soviet artificial earth satellites, the cosmic ship-satellites, and cosmic rockets were equipped with apparatus for studying the radiation near the earth and in cosmic space (Figures 131-133). The principles of construction of this apparatus are given in Chapter II.

The measurements on the second sputnik were made with the aid of charged-particle counters.

The quantity of matter surrounding the counters amounted, on the average, to 10 $g/cm^2$.

During the flight of the sputnik over the territory of the Soviet Union the measurements were made on direct and inverse windings. The flying altitude of the sputnik on direct windings was 225-240 km, while on inverse windings it increased from 350 to 700 km, as the latitude decreased from 65° to 40° North Latitude. The measurements of these altitudes made it possible to ascertain the dependence of the intensity of the primary cosmic radiation on the altitude and on the geographic latitude and longitude.

Figure 134 shows the dependence of altitude of the ratio between the cosmic-ray intensity on inverse windings and the intensity on direct windings in one and the same geographic points. From the diagram it is clear that at medium latitudes, as the altitude changes from 225 to 700 km, the cosmic-radiation intensity increases approximately 40%. This situation can be interpreted in various ways. It may be that the increase in intensity is due to a decrease in the shielding action of the earth and the effect of its magnetic field, which prevents cosmic radiation from penetrating to the earth. Nor is it impossible that the increase in the cosmic-radiation intensity is related to the beginning of penetration into the radiation zone.

Measurements of cosmic-ray intensity with respect to latitudes are of great interest in view of the fact that they enable us to obtain new data

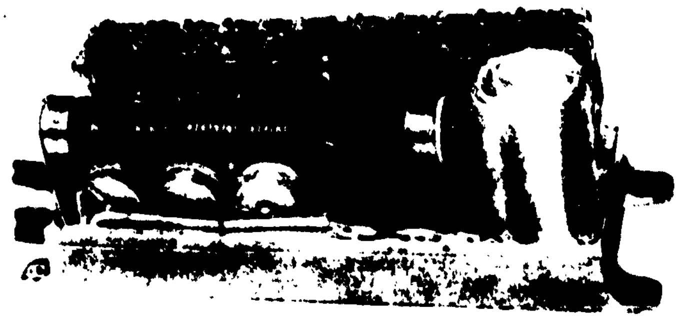

Figure 131. Cosmic-Ray Counter Installed on the Second Sputnik

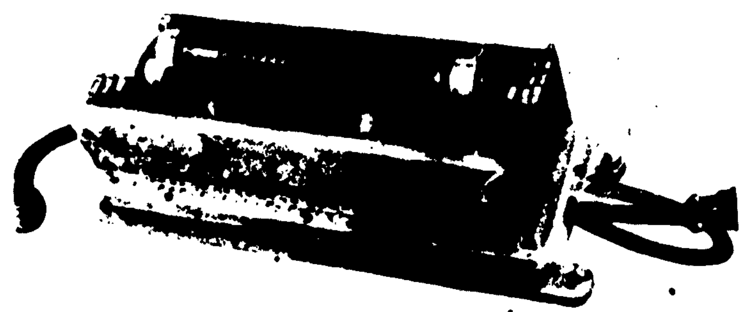

Figure 132. Cosmic-Ray Counter Installed on the Third Sputnik

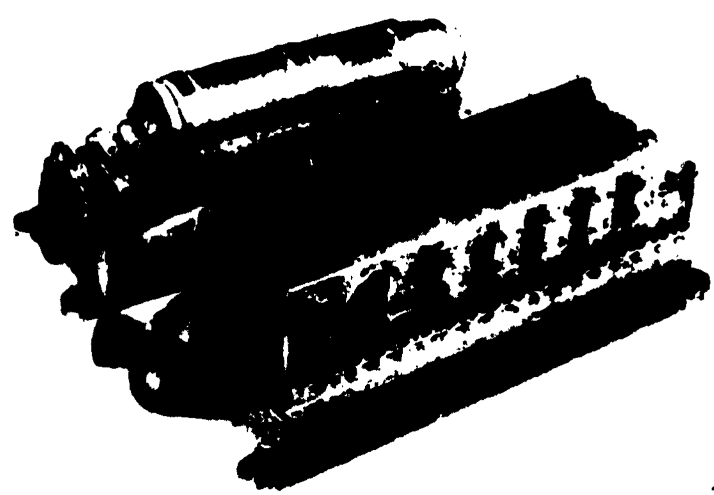

Figure 133. Luminescence Counter Installed on the Third Counter

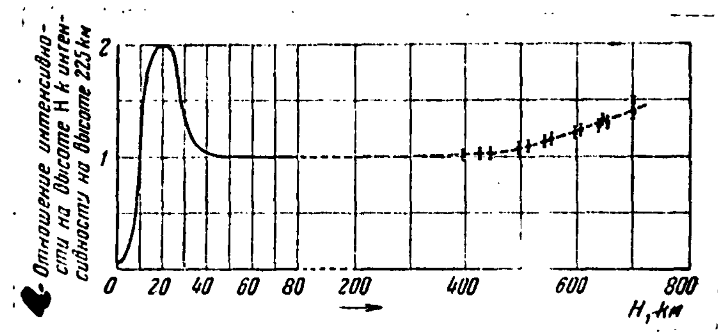


Figure 134. The Dependence on Altitude of the Ratio Between the Cosmic-Ray Intensities on Direct and Inverse Windings. a) the ratio between the intensity at altitude H and the intensity at the altitude 225 km.

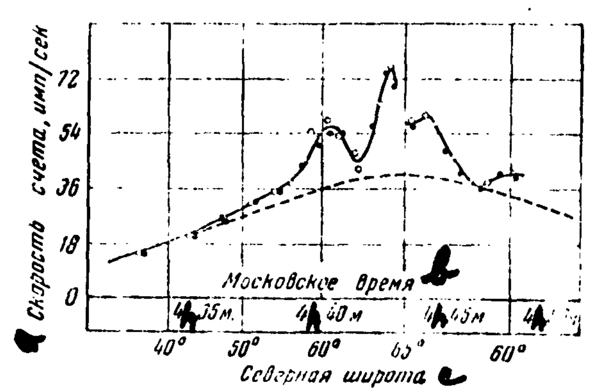


Figure 135. Recording of High Intensity at High Latitudes on the Second Artificial Satellite, November 7, 1957 a) counting rate, pulse/sec; b) Moscow time; c) North Latitude.

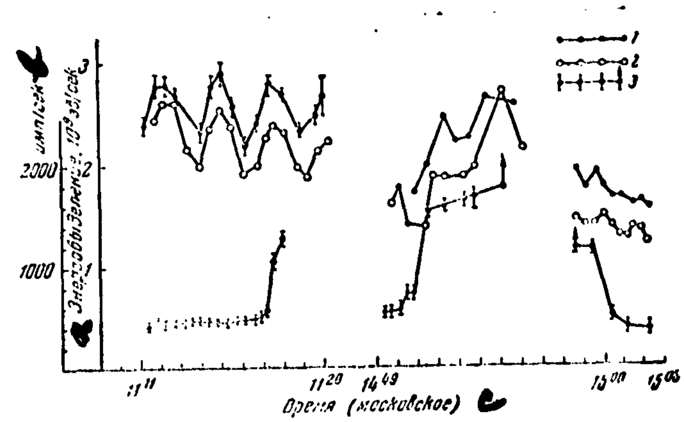


Figure 136. Characteristic Recording of the Counting Intensity and the Ionisation According to the Data of a Luminescence Counter 19 May 1958.
1) ionization according to measurements of dynode current;
2) ionization according to measurements of plate current;
3) counting intensity
a) energy release, $10^9$ ev/sec; b) pulse/sec; c) time (Moscow)

concerning the earth's magnetic field at great distances from the surface of the earth.

The lines of constant cosmic-radiation intensity (isocosms) plotted from the measurements on the second sputnik failed to coincide with the geomagnetic parallels. This attests to the fact that the characteristics of the magnetic field at high altitudes differ from those obtained on the basis of magnetic measurements on the surface of the earth.

The measurements on the second sputnik recorded short-period variations (fluctuations) in cosmic-radiation intensity apparently related to the state of the interplanetary medium near the earth. In one case a sharp increase (50%) in the number of cosmic-radiation particles was noted (Figure 135), at the same time that stations on the earth detected no noticeable changes in radiation intensity. It may be that this increase was caused by an intrusion of the sputnik into high-energy electron fluxes (related to corpuscular solar radiation) or by the generation on the sun of low-energy cosmic rays, which are strongly absorbed by the earth's atmosphere.

The third Soviet artificial earth satellite was equipped with a considerably more sensitive apparatus, a luminescence counter (see Figure 133).

The counter consists of a cylindrical crystal of sodium iodide and a photomultiplier with a photocathode. This device was used to measure:

the event-counting rate, when the pulse corresponded to an energy release in the crystal of more than 35 kev;

the plate current of the photomultiplier;

the current of the intermediate dynode.

The last two parameters characterize the total energy release in the crystal per unit time, which makes it possible to determine the total ionization in the crystal. The operation of this counter and its circuitry are described in Chapter II.

The measurement data from the luminescence counter were transmitted to earth by means of a "Mayak" radio transmitter on a frequency of 20 Mc; this transmitter operated uninterruptedly during the flying time of the sputnik. The "Mayak" transmitter transmitted information by changing the length of the telegraph pulses, the configuration of which is shown in Figure 83.

From the readings of the counter it was established that in all cases without exception, when the sputnik entered the field of geomagnetic latitudes 55–65°, either in the Northern or the Southern Hemisphere, a sharp increase was noted in the intensity of the X-ray radiation, which is created by electrons bombarding the housing of the sputnik. The energy of these electrons is about 100 kev, or less, while their flux has a value of $10^3$–$10^4$ particles/cm · sec · ster.

Figure 136 shows a characteristic recording of the counting intensity and the ionization. The lower represents the counting rate, while the upper curves represent the ionization according to measurements of the dynode and plate currents. The results of the measurements indicate that the recorded ionization value is several times greater than the ionization caused by cosmic rays. This is also borne out by the readings of the dynode and plate currents, the difference in the measurements of which is not great.

Figure 137 shows a geographical map, on which the places where the sputnik enters the zone of high intensity are indicated by dots, while the places where the sputnik emerges from this zone are indicated by crosses. The dashed line indicates the geomagnetic parallel. As can be seen from the diagram, the zone of high intensity is not symmetrical with respect to the magnetic pole.

In the tests we are describing it was also established that the radiation intensity increases with altitude. This fact indicates that in the zone of polar auroras there occurs an accumulation of charged particles, which oscillate along the lines of force of the magnetic field.

Thus the tests on the third artificial earth satellite prove beyond a doubt the presence of an intense radiation zone, which is called the outer radiation belt around the earth. From this it follows that the earth's magnetic field is for low-energy charged particles a unique trap, in which the particles can move along practically closed trajectories for a very lone time.

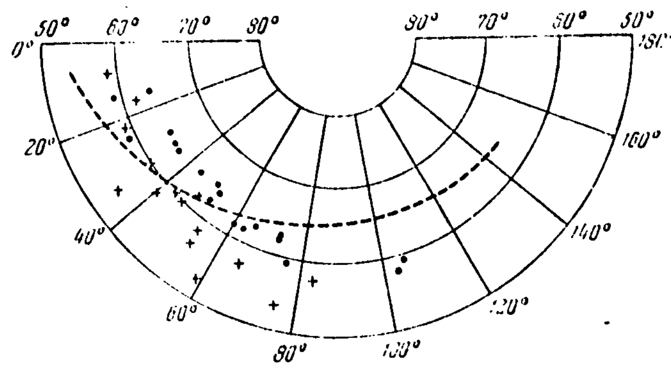


Figure 137. Map Showing the Places where the Third Sputnik Entered (dots) and Emerged from (crosses) the Outer Zone on the Low-Latitude Side.

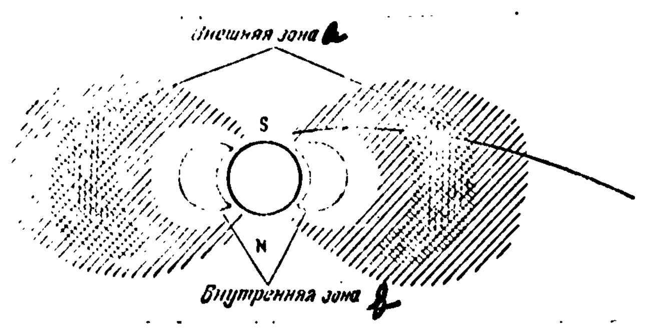


Figure 138. Configuration of the High-Radiation Zones Surrounding the Earth. Solid Line is the Trajectory of Motion of the Cosmic Rocket.
a) outer zone; b) inner zone

The conditions for accumulation of particles are not fulfilled at latitudes greater than 65°, and therefore the regions adjacent to the poles are free of radiation.

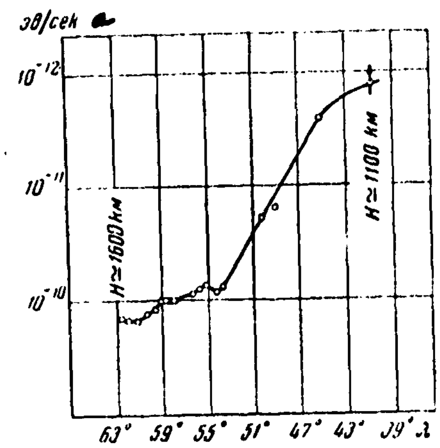


**Figure 139. Characteristic Recording Made by a Luminescence Counter in the Southern Hemisphere.**
**a) ev/sec**

In addition to the outer radiation zone around the earth, there exists an inner radiation zone, located in the region of the equator at an altitude of ~1000–2000 km (Figure 138).

Detailed data concerning this zone were obtained with the aid of the third sputnik. In Figure 139 we have reproduced one of the recordings of readings of the counter; it was obtained in the Southern Hemisphere at the altitudes 1600–1100 km.

The measurement data indicate that as the sputnik moves toward the equator the radiation intensity increases sharply, in spite of the decrease in the altitude of the sputnik from 1600 to 1100 km. The latitude plays a significant role in this process. It was found that charged particles of the inner zone fill the region from 35° south geomagnetic latitude to 35° north geomagnetic latitude at an altitude of approximately 1000 km. The altitude of the lower boundary of the inner zone was found to be different in the Eastern and Western Hemispheres: in the Eastern 1500 km; in the Western 500 km;. This is due to the shift of the magnetic dipole relative to the center of the earth. An analysis of the data showed that protons with an energy of the order of 100 million ev are most characteristic of the inner zone.

Further study of the outer radiation zone took place during the flights of Soviet cosmic rockets.

The first cosmic rocket recorded the radiation intensity near the earth, and the cosmic radiation with the aid of two Geiger counters and two scintillation counters.

The first device containing a scintillation counter was similar to the device installed on the third sputnik. It was used to measure the number of events with the energy threshholds: I - 45 kev; II - 450 kev; III - 4.5 Mev; IV - total ionization.

Both Geiger counters and the first scintillation counter were located inside an aluminum casing 1 g/cm$^2$ thick. Approximately 20% of the total solid angle was shielded by material of the order of 10 g/cm$^2$. The second scintillation counter was located outside the shielding casing. The scintillator, which was 0.3 g/cm$^2$ thick, was covered on the free-space side with aluminum foil 1.9 mg/cm$^2$. This device recorded only the total ionization in the crystal. The measurements were made at distances of 8 to 150 thousand km from the center of the earth. As a result of the measurements, the spatial distribution of the outer zone was obtained, and the composition of the radiation in the outer zone was studied in more detail. It was found that the effective energy in the regions of the maximum is approximately 25 kev, while on the boundary of the zone it is approximately 50 kev.

A comparison of the readings of all the instruments installed on the first cosmic rocket enables us to establish that the maximum intensity is reached at a distance of 26 thousand km from the center of the earth. At a distance of 55 thousand km the radiation intensity is practically equal to zero (in relation to the background of constant cosmic radiation). In addition to depending on the distance, the radiation intensity is determined to a large extent by the magnetic line of force on which the measurement is made. It was found

that the particle flux is not unidirectional. The particles oscillate along the magnetic lines of force from one hemisphere to the other, undergoing total reflection upon approaching the earth, according to the law:

$$\frac{\sin^2\theta}{H} = \text{const}, \qquad (4.20)$$

where $\theta$ is the angle between the velocity vector of the particle and the magnetic-field vector at the given point of the trajectory.

Thus the outer zone, according to the data of the first cosmic rocket, must be conceived of as being located in the space between the magnetic lines of force 55° and 67°. The maximum intensity is observed on the line of force 62° (Figure 140). Beyond the limits of the outer zone (66 to 150 thousand km) the first cosmic rocket measured the primary cosmic radiation, which is unaffected by the earth's magnetic field at such distances. This means that either the earth's magnetic field "disappears" at distances of 10 earth radii, or that there are no particles in the cosmos, which could be deflected by a magnetic field of the order of $3 \cdot 10^{-4}$ oe.

The flux primary cosmic rays amounts to $2.3 \pm 0.1$ particles/$cm^2 \cdot$ sec. The photon intensity in the interval $45 \pm 450$ kev is $3.2 \pm 0.1$ photons/$cm^2 \cdot$ sec and is $1 \pm 0.1$ photons/$cm^2 \cdot$ sec in the interval $450 \pm 4500$ kev. The energy flux of the photons is very small and makes practically no contribution to ionization (see Figure 140).

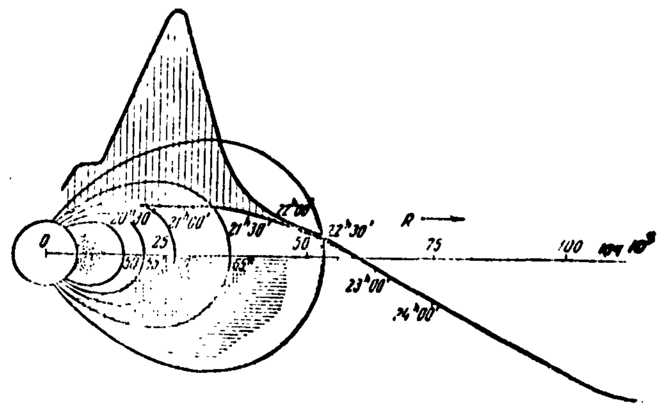


Figure 140. Trajectory of motion of the first Soviet Cosmic Rocket in geomagnetic coordinates. The Moscow flying time and the radiation intensity (vertical lines standing on the trajectory) at a given point (according to ionization measurements in a sodium-iodide crystal) are indicated along the trajectory. The magnetic lines of force intersecting the surface of the earth at the geomagnetic latitudes 50, 55, 60, 65 and 70° are shown. The outer belt is denoted by hatching, the inner belt by dots.

Various devices were installed in the second cosmic rocket, their use made it possible to conduct investigations even deeper into the outer zone, and also studies in which radiation bands were discovered aroung the Moon. The apparatus consisted of gas-discharge and scintillation counters. They were installed inside as well as outside the housing.

Inside the housing one scintillation counter was installed to record complete ionization and the pulse-count rate (the pulses corresponded to energy liberation in a crystal: I $\gtrsim$ 60 Kev, II $\gtrsim$ 600 Kev, III $\gtrsim$ 3.5 Mev, and two gas-discharge counters with additional shielding; one counter had a copper shield 1.5 mm thick, and the other a lead and an aluminum shield, 3 and 1 mm thick respectively. All three instruments were located in a shell with a thickness of 1 g/cm$^2$ aluminum. In addition, about 20% of the total solid angle was covered with a substance 10 g/cm$^2$ thick.

Two scintillation counters were installed outside of the housing. One recorded complete ionization and was closed from the free-space side with aluminum with a thickness of 1.2 mg/cm$^2$, the other recorded complete ionization and the pulse-rate count corresponding to the energy liberation: I $\gtrsim$ 45 Kev, II $\gtrsim$ 450 Kev. The crystal of this counter was shielded with 1 g/cm$^2$ aluminum, and only about 5% of the total solid angle was covered by a greater quantity of the substance ($\sim$10 g/cm$^2$). Three gas-discharge counters were also mounted outside the housing. The first was shielded with 3 mm of lead and 1 mm of aluminum having a window with an area of 2.8 cm$^2$, the second had the same kind of shield, but the window had an area of 1.6 cm$^2$, shielded with copper foil 0.2 mm thick, the third had the same shielding and a window with an area of 1.6 cm$^2$, shielded with copper foil 0.5 mm thick. In addition, all three windows were shielded with aluminum foil 0.2 mm thick. The wall thickness of all three counters was 50 mg/cm$^2$ of stainless steel. The second and third gas-discharge counters, mounted outside the housing, operated only in the high-intensity zone. After leaving the high-intensity zone, the telemetry channels

transmitting data from the gas-discharge counters were switched to the transmission of information from the scintillation counters.

Putting this apparatus on board the second cosmic rocket made it possible to obtain new information on the three-dimensional location of the outer radiation zone.

Figure 141 shows the position of the maximums of the high-intensity zone according to data from the first and second cosmic rockets. As has already been stated, the maximum of the outer radiation zone of the Second of January was observed at a distance of 27,000 km on a line of force of 62°. On the 12th of September the maximum was observed at a distance of 17,000 km from the center of the Earth on a line of force of 59°. The reasons for this may be varied. In the first place, the different trajectory positions on the 2nd of January and the 12th of September relative to the direction on the Sun, which could have caused a systematic deformation of the Earth's magnetic field; in the second place, the deformation of the belt could have been due to the varying nature of the corpuscular currents and therefore, due to the varying nature of injection of particles into the radiation zone. The difference in the spectrum of particles recorded on the 2nd of January and the 12th of September speaks in favor of the latter case.

The results of measurements of the particle composition of the outer belt on the second cosmic rocket support the data from the first cosmic rocket that particles with a path of several $g/cm^2$ are absent in the outer radiation belt. Essentially new data were obtained from the readings of gas-discharge counters, installed inside the housing and shielded with additional filters of copper and lead. Both counters recorded photons with energy greater than 400 Kev. An analysis of the readings of the gas-discharge counters allow it to be assumed that there exist two separate groups of particles; electrons with energies of 20 Kev and electrons with an energy of 2 Mev (or protons with an energy

of 10 Mev). Apparently the formation mechanisms of both groups are essentially different.

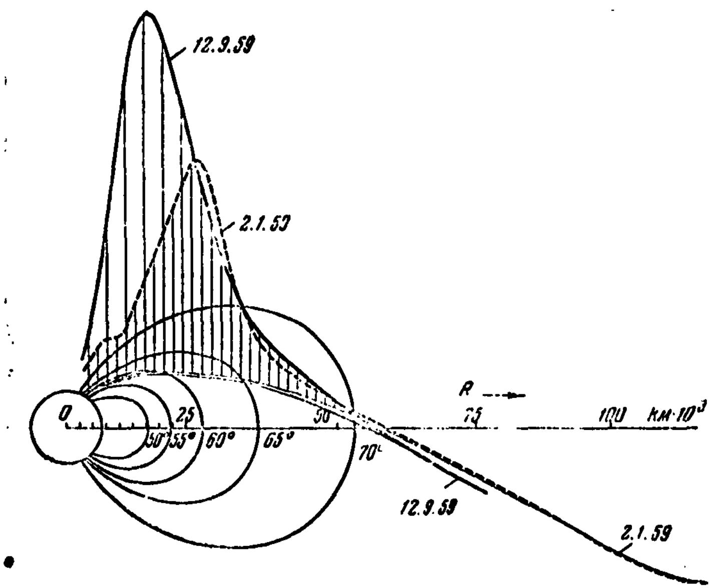


Figure 141. Maximum of the High-Intensity Zone According to Data from the First and Second Cosmic Rockets.

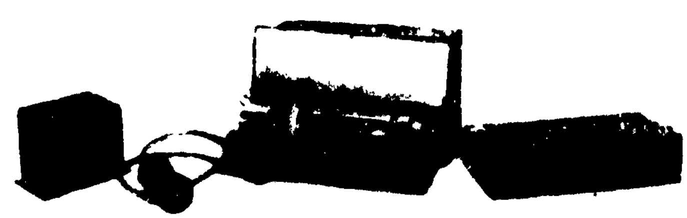

Figure 142. An Instrument for Measuring the Quantity of Heavy Nuclei in Cosmic Radiation.

One of the most important problems in launching the rocket on the 12th of September was the discovery of the Moon's radiation belts. The result is negative: In approaching the Moon up to a distance of 1000 km from its surface, the increase in radiation intensity of 10% of the cosmic background is not detected.

Thus, it may be assumed that for all practical purposes there are no Lunar radiation belts.

Apparatus for measuring heavy nuclei in primary cosmic radiation was



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installed on the third artificial Earth satellite and the cosmic rockets. A Cerenkov counter served as the sensitive element of the instrument; it consisted of a plexiglass detector and a photomultiplier. One of these instruments is shown in Figure 142.

The determination of the charge of a particle was made in the Cerenkov counter by measuring the intensity of luminescence, which is proportional to the square of the charge, (See Chapter II).

The instrument installed on the third artificial satellite recorded the nuclei with kinetic energy greater than $3 \cdot 10^8$ ev/nucleon. The instrument was adjusted to record two groups of nuclei: with a charge greater than 15 to 20 and with a charge greater than 30 to 40. Processing of the data on the operation of the instrument for 9 days showed that $1.22 \pm 0.8$ particles with $Z > 15$ to 20 per minute passed through the instrument on the average. Only one case of the wear of a channel, tuned to $Z > 30$ to 40, was noted in the course of the 9 days. This evaluation showed that the maximum number of nuclei with $Z$ 30 to 40 passing through the Cerenkov counter did not exceed 1 to 3. Hence it follows that a stream of nuclei with $Z > 30$ to 40 makes up not more than 0.03% of a stream of nuclei with $Z > 15$ to 20. Thus, it should be assumed that the stream of heavy nuclei is small, and that the indication of the existence of a stream of nuclei with $Z > 30$, comparable with a stream of nuclei of the iron group in magnitude, is not supported.

Cerenkov counters for recording aplha particles and nuclei with $Z > 5$ and $Z > 15$ were installed on the second cosmic rocket. The counter for recording alpha particles was located outside the hermetically sealed housing, the others were inside it. The thickness of the shell of the housing did not exceed 1 g/cm$^2$ of aluminum. The counters recorded nuclei with a total energy greater than $1.3 \cdot 10^9$ ev/nucleon. The channels along which the information on nucleus recording was transmitted were designed for determined threshold values of

energy, but counts of alpha particles and nuclei with $Z > 5$ and $Z > 15$ can be made by them.

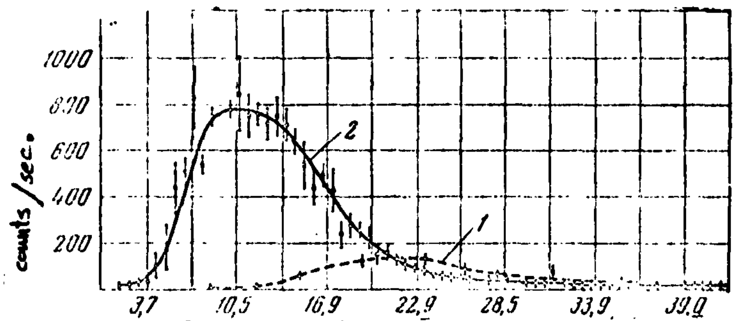
In addition to the nucleus-count channels, there was a channel for recording the intensity of all charged particles in the radiation belts, the so-called radiation indicator. The radiation indicator, besides recording low-energy charged particles (electrons with an energy of 15 to 20 Kev) creating X-ray radiation in the shell of the housing to which the photomultiplier of the Cerenkov counters were sensitive, could also record electrons passing through the shell of the housing and having a kinetic energy greater than 2 Mev. Protons and nuclei with a total energy greater than $1.3 \cdot 10^9$ ev/nucleon were recorded by Cerenkov radiation.

Figure 143 shows the course of radiation intensity recorded by the radiation indicator as a function of distance. Curve 1 was obtained during the flight of the first cosmic rocket and attests to the presence of a maximum radiation intensity at distance of 22,000 km from the Earth's surface; curve 2 was obtained during the flight of the second cosmic rocket and attests to the presence of maximum radiation intensity at a distance of 10,000 km from the Earth's surface. From these curves, as well as from a comparison of data from luminescence counters (see above), it is apparent that the maximum radiation intensity, as well as the entire radiation belt obtained during the flight of the second cosmic rocket, was shifted toward the Earth in comparison with the maximum recorded during the flight of the first cosmic rocket.

Beyond the outer radiation belt the radiation indicator recorded only protons of primary cosmic rays. The stream recorded equalled 2 to 4 particles/$cm^2 \cdot sec$.

In the vicinity of the Moon and during the approach to it, the indicator did not detect a noticeable increase in intensity. The information from the channels recording -particles and nuclei with $Z > 5$ and $Z > 15$ indicates

that the mean values of $\alpha$-particle-streams and the aforementioned groups of nuclei do not vary with distance at long distances from the Earth. An approximate stream value of 140 ± 10 to 150 ± 10 particles/m$^2$·sec.sterad was obtained for $\alpha$-particles; 10.0 ± 0.3 particles/m$^2$·sec·sterad for $Z > 5$; and 0.37 ± 0.06 particles/m$^2$·sec·sterad for $Z > 15$, at determined values of the geometric factor of the Cerenkov counters[1].



Distance from Earth's Surface, Km · $10^3$

Figure 143. The Course of the Radiation Intensity Recorded by a Cerenkov Counter During the First and Second Cosmic Rocket Flight.

During the flight of the second cosmic rocket an interesting phenomenon was discovered: At 1127 UT, 12 September 1959, the number of nuclei with $Z \geq 15$ entering the counters increased by a factor of 11.8 (11.8 ± 0.7) in comparison with average intensity. This increase lasted about 17 minutes. At the same time, the number of nuclei with $Z \geq 2$ and $Z \geq 5$ increased by factors of 1.3 ± 0.1 and 1.5 ± 0.3 respectively.

An analysis of this phenomenon showed that it was connected with processes taking place on the Sun: at a time interval closely coinciding with increase in intensity of the nuclear component, two chromospheric eruptions and also an outburst of radio emission were recorded at stations on Earth.

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[1] The geometric factor in this case is the value, having dimension [cm$^2$·sterad], the product of which by the value of the stream [particles/cm$^2$·sec·sterad] equals the number of counter readings per second.

A comparison of these data leads to the idea that on the Sun processes apparently take place in which nuclei are accelerated to energies exceeding $1.5 \cdot 10^9$ev/nucleon.

Essentially new results in the study of cosmic rays were obtained during the flights of the second and third satellites.

As is known, the orbit of the satellites was at an altitude of 200 to 300 km. As a result of measurements, a chart of the distribution of intensity over the entire Earth was obtained. It follows from this chart that near the equator the intensity of radiation is comparatively low. High-energy cosmic-ray particles occur at the equator. Moving from the equator to higher latitude, radiation increases. This takes place because far off from the equator, not only high-energy particles, but also cosmic-ray particles of low energy reach the Earth from space, particles of low energy than is necessary to reach the equator.

At latitudes exceeding 50° an increased number of electrons is observed. This is the beginning of the outer radiation belt of the Earth. When approaching the Earth's magnetic pole, the radiation intensity again drops; particles of the radiation belt do not penetrate to here, only cosmic rays.

Finally, this chart indicates a considerable increase in radiation intensity in the region of the South Atlantic. This phenomenon is apparently connected with the existence of anomalities there of the Earth's field.

<u>Studies of Interplanetary Gas Using Ion Traps</u>.

Experiments were run on all three Soviet cosmic rockets to study ionized gas in interplanetary space and in the uppermost layers of the Earth's atmosphere. Special instruments were developed for this; three-electrode charged-particle traps.

On the rocket launched 12 September 1959, four traps with various external-grid potentials were set up: +15 v, -5 v and -10 v. Positive current could be

caused only by ions, wherein their energy could be determined by comparison of the readings of all traps. If all traps simultaneously recorded negative current, this meant that it was caused by electron streams. In this case the energy of each electron exceeded 200 ev because only those electrons could pass through the decelerating potential applied to the internal grid of the trap.

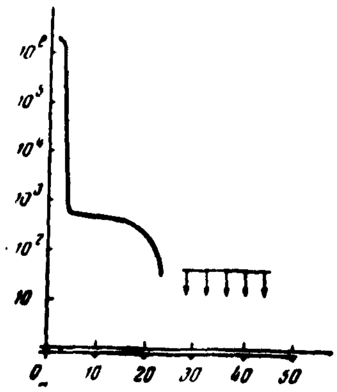
As a result of the measurements conducted on the three Soviet rockets using traps, a great deal of experimental material was obtained. Only during the flight of the Soviet rocket in September, 1959, there were about 12,000 trap-current measurements transmitted back to Earth.

Let us examine the fundamental scientific results of the investigations using traps. Figure 144 shows a graph of ion concentration versus the distance from the Earth's surface in kilometers. Two fundamental conclusions can be made from this.

1) The Earth is surrounded by a very expansive and highly rarefied atmosphere of ionized gas. This atmosphere may be rightfully called the "Earth's corona" or "geocorona". The ion concentration in the geocorona is on the order of several hundred positively charged particles per $cm^3$. For comparison let us say that the ion concentration of the Earth's atmosphere at an altitude of 300 km reaches 1 to 2 million per $cm^3$. And the concentration of molecules in the atmosphere at the Earth's surface is expressed by a 20-digit number.

From the nature of the change in ion concentration with distance from the Earth's surface it may be concluded that the geocorona is composed of hydrogen. The geocorona can be detected up to a distance of 22,000 km from the Earth's surface. There is, however, reason to assume that the extent of the geocorona varies. It may be a function of several things, chiefly of solar activity.

2) In interplanetary space at distances exceeding 22,000 km from the Earth's surface, there was no measureable concentration of ionized gas. Hence,




Distance from the Earth's surface, km $10^3$.

Figure 144. Ion Concentration Versus Distance.

assuming that the experiment were accurate, it may be concluded that if there is ionised gas in interplanetary space, its concentration is surely lower than 100 ions/cm$^3$. According to existing circumstantial data based on analysis of trap currents, the concentration of ionised interplanetary gas must be even considerably lower than this.

3) At distances from the Earth's surface from 45 to 80 thousand km, streams of electrons with energies exceeding 200 ev and with current density $N \sim 10^8$ particles per cm$^2$ per second were discovered. These streams attest to the existence of an outermost belt of charged particles surrounding the Earth, particles of comparatively low energy and located beyond the radiation belts.

Finally, magnetic measurements conducted on American space vehicles indicated that in the region of space where the outermost belt is located, there exists a current which distorts the Earth's magnetic field. This current is apparently a drifting current of electrons in the outermost belt, caused by nonuniformity of the magnetic field.

## Measurement of Magnetic Field Near Earth and Moon

A recording manometer with magnetically saturated pickups was installed on the third Soviet satellite for geomagnetic measurements in and outside ionosphere of the Earth. The operation of such a magnetometer is described in Chapter 2.

The magnetometer developed for the third satellite (Fig. 145) insured measurements of the earth field intensity with a high accuracy, on any magnetic latitude and with any orientation of satellite, and was fully automatic.

Fig. 145. Magnetometer Installed on Third Satellite.

The magnetometer had also a special appliance which made it possible to obtain data on changing space orientation of the satellite (relative to the vector of magnetic field), and on variation in the nature of satellite revolution. The measuring range of the instrument constituted 48,000$\gamma$. The performance rate of servosystem was equal to 40 - 45°/sec; the mean angle of zero point was about 2$\gamma$ per hour. The zero point of the magnetometer was determined by comparison with proton magnetometer. The maximum value of magnetic deviation amounted to approximately 3,000$\gamma$. Measurements were conducted on altitudes of 250 - 750 km, i. e., in the region below and above the maximum of ionization



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of the $F_2$ layer. Magnetic error, the period of which coincides with the satellite precession period, was fixed on recordings. If the satellite orientation is known, a major portion of this error can be eliminated.

Magnetic investigations on the third Soviet satellite furnish a convincing proof for the presence of ionospheric sources which cause variations associated with the earth field disturbance. Analysis of the obtained material indicates the presence of short, fast changes in the earth field, coinciding in time with the passage of the satellite through the $F_2$ layer of ionosphere (Fig. 146). At the same time, 20 cases of short (5 to 8 sec) negative and positive peaks of the earth field variation were detected. They can be considered due to space inhomogeneities in ionospheric current systems of local character, intersected by the satellite.

Fig. 146. Characteristic Recording of Short Earth Field Variation.

New data were obtained during the examination of constant magnetic earth field, particularly when the satellite was flying above the Easter Siberian anomaly, the so-called Asiatic maximum of intensity of the geomagnetic field. As it is demonstrated by the analysis of obtained data, the magnitude of this anomaly attenuates with the altitudes very slowly.

Basing on indications of two potentiometric pickups, characterizing angular displacements of the satellite body, it is possible to obtain a numerical value of the relative variation in the space position of the satellite for any moment

of time. It follows from these data that the satellite was performing a precession motion with the period of $T = 136$ sec. Apart from that, the satellite was revolving around its own axis at the rate of $0.35^{\circ}$/sec. These results are very important for the interpretation of indications of other instruments installed on the satellite.

On the first and second satellites were performed magnetic measurements at distances of several terrestrial radii, as well as in the environs of the Moon. These measurements are highly important for geophysical researches, particularly for the experimental verification of existing theories of magnetic storms and aurora polaris.

Measurements on the first space rocket were conducted by means of a three-component magnetometer with magnetically saturated pickups of even harmonics (refer to Chapter II). General view of such a pickup is shown in Fig. 147. The measuring range on each pickup was equal to $\pm 3000\ \gamma$. The magnetometer's drift from zero did not exceed $20\gamma$ in 24 hours of continuous operation. The sensitivity of each channel of the magnetometer equalled 600 v. The total deviation from magnetic components of the rocket did not surpass $70\gamma$ and was known for each pickup.

Fig. 147. Pickup of Magnetometer Installed on First Space Rocket.

Fig. 148 shows the experimental curve for intensity of the terrestrial magnetic field as a function of distance, as well as the theoretical curve of the dipole field.

The comparison of measured and calculated field values shows that measured values



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differ substantially from calculated ones in the flight phase at 15 to 30 thousand km from the earth center. The disagreement between the true field and the dipole field equals 300 $\gamma$ even at the distance of about 14,700 km. Here is observed an abrupt decrement in the terrestrial magnetic field with distanse. The difference between the true field and that calculated from the earth potential attains its maximum value at the distance of 19 to 20 thousand km and equals 800 $\gamma$.

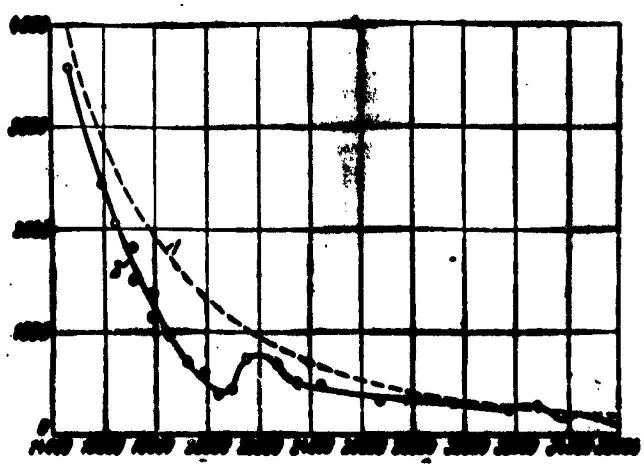


Fig. 148. Variation in Magnetic Earth Field with Altitude.

1 - calculated values; 2 - measured values.

The result obtained has a great scientific significance. It permits to assert that the magnetic earth field at distances of 3 to 4 terrestrial radii is determined not only by values calculated from the magnetic earth potential, but depends also on external sources.

On the second space rocket, launched to the Moon on 12 September 1959, were installed more sensitive magnetometers of the same type as on the first rocket. The sensitivity of magnetometers constituted 115 $\gamma$/v of telemetric system. At the same time, the measuring range was reduced which caused the measurements of the terrestrial magnetic field on the second space rocket to start approximately from the altitude of 18,000 km from the earth center.

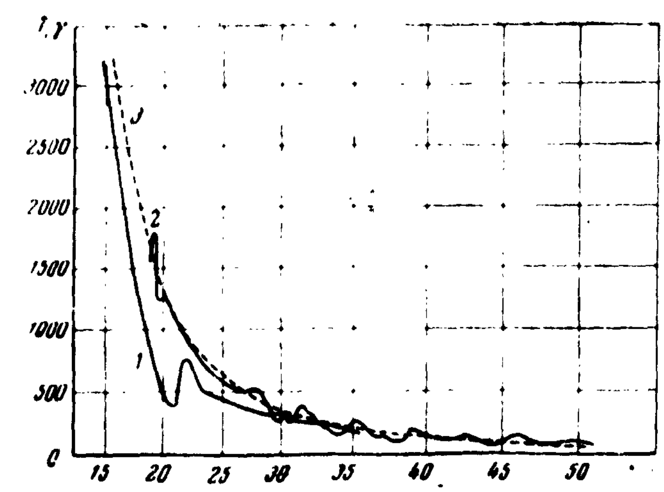


Figure 149. Values of magnetic terrestrial magnetic field intensity obtained by various methods:

1- measured values (1st cosmic rocket ; 2- measured values (2nd cosmic rocket); 3 - computed values

Magnetic measurements on the second cosmic rocket were carried out more frequently, than on the first one. Reading of magnetometers was done every 3-6 seconds, and then periodically repeated within approximately one half of a minute.

On fig.149 are shown curves, obtained on the first and second cosmic rockets. As already pointed out before , the measurements on the second cosmic rocket revealed instability of the outer part of the radiation zone: during flight on September 12 1959, its maximum was situated closer to Earth, than during the time of flight on January 2, 1959. Because of this the reduction in geomagnetic field intensity during measurements on the second cosmic rocket has not been detected, because the anticipated magnetic effects were in the zone of the field, where the intensity value was beyond the limits of magnetometer measurements. At present time there is a great discussion going on between astrophysicists and geophysicists regarding the nature of radiation bands and the causes of magnetism of radiation belts.

The flight of the second cosmic rocket to the Moon and the magnetic measurements in direct vicinity from the Moon discovered no noticeable increase in magnetic field intensity. Magnetograms, obtained from the near-lunar section, were subjected to thorough statistical analysis . This allowed to make a conclusion; that if at the Moon would exist a magnetic field of more than 100 gammas, then it would have been detected. Consequently, it can be stated, that there is no magnetic field at the Moon, which by its magnitude would exceed the measurement errors. That is why efforts to explain certain geophysical phenomena by the effect of the lunar magnetic field should now be considered as absolutely baseless.

Investigation of Micrometeors

To investigate meteoric particles on the third man-made Earth satellite and on cosmic rockets was installed an apparatus, consisting of ballistic Piezo-feelers (from ammorous phosphate) and amplifier-converter (see chapter 2). General view of the apparatus is shown in Fig. 150. The apparatus secured the registration of the number of particle impacts, and registration of their energy as well, energy measured by the pulse magnitude of the material of the sensing device, ejected during the explosion of a meteroite particle on its surface:

$$J = AE_{\circ}, \qquad (4.21)$$

See page 153 for Figure 150

Fig.150. Apparatus for studying meteoric particles , installed on third satellite.

where L = "ejection " pulse; $E_{\circ}$ = kinetic energy of particles; A = proportionality coefficient.

If the average velocity of the particle is accepted in this case at 40 km/sec, it is then possible to determine the mass of the recorded particles.

On the third man-made Earth satellite were set up collision feelers with total area of 3410 $cm^2$, including the body of the feeler. Distribution of signals by amplitude was done by converting the signals from the amplifier part into a computing system of each amplitude range after a corresponding accumulation of a number of impacts.

On the first and second cosmic rockets were mounted feelers with general area of 0.2 $m^2$, and on the third one - 0.1 $m^2$.

The apparatus mounted on the first cosmic rocket was calibrated for registartion of particles with masses: I - $2.5 \cdot 10^{-9}$ - $1.5 \cdot 10^{-8}$ g; II - $1.5 \cdot 10^{-8}$ - $2 \cdot 10^{-7}$ g; III $> 2 \cdot 10^{-7}$ g.

On second and third cosmic rockets there was no accumulation of impacts,and

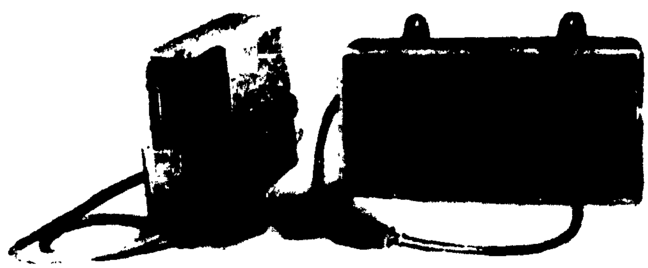

Fig. 150. Apparatus for studying meteoric particles, installed on third satellite.



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each impact was recorded separately. The apparatus installed on the second cosmic rocket was calibrated for recording particles with masses: I = $2 \cdot 10^{-9}$ – $6 \cdot 10^{-9}$ g; II = $6 \cdot 10^{-9}$ – $1.5 \cdot 10^{-8}$ g; III $>$ $1.5 \cdot 10^{-8}$ g, and on the third cosmic rocket: I = $10^{-9}$ – $3 \cdot 10^{-9}$ g; II = $3 \cdot 10^{-9}$ – $8 \cdot 10^{-9}$ g; III = $>$ $8 \cdot 10^{-9}$ g.

The measurement results from the third man-made Earth satellite and cosmic rockets are listed in table 34.

By examining the mentioned table it is possible to arrive at certain conclusions:

1. Density of meteoric matter in the periphery of the Earth is not constant - it changes in space.

2. Maximum number of recorded impacts belongs to particles with masses ranging from $(3 \div 8)10^{-9}$ to $(2 \div 3)10^{-8}$ g and smaller.

These results show that meteoric and micrometeoric danger is small.

TABLE 34

Results of measuring on the third man-made satellite and cosmic rockets

| Cosmic unit | Date | Mass of particles (at V = 40 km/sec) | Number of particle impacts per 1 cm² per sec. |
|---|---|---|---|
| THIRD SATELLITE | 15/5—58<br>16—17/5—58<br>18—26/5—58 | $8 \cdot 10^{-9}$—$2{,}7 \cdot 10^{-8}$ | 4—11<br>$5 \cdot 10^{-4}$<br>$< 10^{-6}$ |
| First Rocket | 2/1—59 | $2{,}5 \cdot 10^{-9}$—$1{,}5 \cdot 10^{-8}$<br>$1{,}5 \cdot 10^{-8}$—$2 \cdot 10^{-7}$<br>$> 2 \cdot 10^{-7}$ | $< 2 \cdot 10^{-3}$<br>$< 5 \cdot 10^{-4}$<br>$< 10^{-4}$ |
| Second rocket | 12/9—59 | $2 \cdot 10^{-9}$—$6 \cdot 10^{-9}$<br>$6 \cdot 10^{-9}$—$1{,}5 \cdot 10^{-8}$<br>$> 1{,}5 \cdot 10^{-8}$ | $< 5 \cdot 10^{-4}$<br>$< 5 \cdot 10^{-4}$<br>$< 9 \cdot 10^{-4}$ |
| Third rocket | 4/10—59<br>18/10—59 | $10^{-9}$—$3 \cdot 10^{-9}$<br>$3 \cdot 10^{-9}$—$8 \cdot 10^{-9}$<br>$> 8 \cdot 10^{-9}$ | $4 \cdot 10^{-4}$<br>$2 \cdot 10^{-3}$<br>$4 \cdot 10^{-4}$ |

Biological investigations

On the second Soviet man-made Earth satellite was carried out a medical-biolog experiment by studying the vitality of a living organism under conditions of cosmic flight, one of the basic characteristics it is the state of weightlessness.

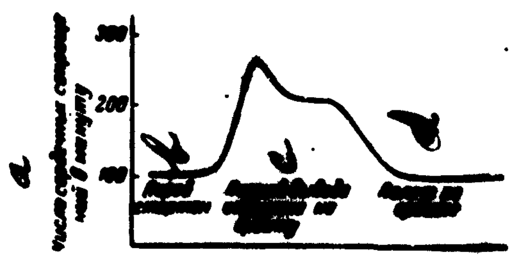

Fig. 151. Frequency curve of cardio contractions of dog Layka at various moments of flight.

a- number of systoles per minute; b- prior to blast off; c- time satellite gets into orbit; d- orbital flight.

For this purpose the satellite was provided with an airtight (pressurized) cabin occupied by an experimental dog - Layka. The cabin was furnished with devices to reproduce conditions necessary for normal existence of an animal during lasting flight, as well as equipment for recording its physiological functions (see Chapter 2).

The data obtained as result of this experiment are of greater scientific value. Of great interest are data on the behavior and condition of the animal on the orbiting section of the satellite, the characteristic of which is the presence of large overloads, as well as vibrations and noise from the operating rocket power plant. The behavior and condition of the animal when the satellite

came into orbit were registered quite fully.

On the basis of information obtained over radio telemetering system it can be established, that up to a define overload magnitude the animal resisted the increase in apparent bodily weight and maintained freedom of movement of head and trunk. Then it appeared to be pressed against the floor of the cabin and some of its noticeable movements have not been recorded. The frequency of the systoles immediately after blast-off increased in comparison with the initial one by 3 times (fig. 151). However an analysis of cardio biocurrents recording (electrocardiogram) indicates the absence of any ill effects (fig. 152). There was a typical chart of more frequent palpitation of the heart, a so-called sinusal tachycardia.

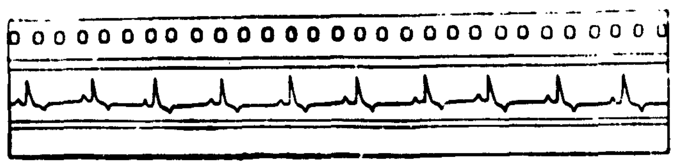

Fig. 152. Electrocardiogram recording of dog Layka in state of weightlessness, obtained on the second satellite.

Toward the end of the orbiting, in spite of increase in overload, the frequency of cardio palpitation began decreasing. The respiration frequency of the animal on the orbiting section was 3-4 times greater than the initial one. As the apparent weight of the animal increased the respiratory movement of the thorax became difficult, as result of which its respiration became more superficial and frequent.

There are bases to assume, that the indicated changes in the state of physiological functions of the animal are connected with the sudden action against the organism of sufficiently strong outer stimuli - overloads, noises, vibrations. Analysis of obtained data and their comparison with results of

preceding laboratory experimentations allow to state that the flight along the orbiting section was endured by the animal with satisfaction.

After the satellite reached orbit the state of weightlessness took place. The body of the animal under these conditions during contraction of extrimity muscles rolled away easily from the flow of the cabin. Judging by the available recordings, these movements were of short duration and smooth. The respiration frequency began dropping. The cardio systoles subsequently began decreasing, coming down to initial magnitude. But the time, within the heart palpitation became normalized, was approximately three times greater, than during laboratory experiments, at which the animal was subjected to the effect of such accelerations, as during orbiting. In all probability this is explained by the influence of the weightlessness state, at which sensitive nermous formations, signalling the position of the body in space, have not experienced sufficient influence of external stimuli, which in turn led to a change in functional state of the nervous system and established a certain extension in the period of frequency normalization in heart palpitation and respiration.

Analysis of the electrocardiogram taken in state of weightlessness, immediately after the satellite began orbiting, revealed certain changes in the configuration of its elements and the continuance of individual intervals. But these changes were of no pathological nature and were connected with the increased functional load during the period, preceding the state of weightlessness. The picture of the ECG (electrocardiogram) reflected timely nervous-reflector displacements in the regulation of cardio activity. Next was ovserved a greater approach of ECG to the one, which is characteristic for the initial state of the animal. The motorial activity of the animal, in spite of the unusual state of weightlessness, was moderate.

Normalization of functional characteristics of blood circulation and respiration under weightlessness conditions attests most obviously that this fact in itself caused no essential and permanent changes in the state of physiological functions of the animal. No definite opinion regarding the effect of cosmic radiation on the experimental animal could be made. No physiological effect of its action has been directly discovered.

The biological experimentation results obtained on the second satellite and on ships-satellites show beyond any doubt that cosmic flight conditions are endured by the animal with satisfaction[1].

The experimentation result positive in this case makes provisions for continuing and expanding the investigations, with the purpose of creating conditions, safe for the health and life of humans in cosmic flight.

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1. About flights of animals on ships-satellites see Chapters VI and VII.

First Photos of the Reverse Side of the Moon

The period of rotation of the Moon around its axis coincides with its period of rotation around the Earth, and that is why the Moon is turned to the Earth always with one and the same side. The presence of so-called lunar librations, i.e. periodic oscillations of the Moon about its center, visible for ground observer, made it possible to investigate and plot on charts 59% of its surface.

Figure 153. Photo of reverse side of the Moon obtained from board the AIS (automatic interplanetary station)



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**Fig.154.Photo of the reverse side of the Moon taken from board of the AIS**

Some lunar formations are situated along the very edge of the visible disk,whereby a part of them is visible only during corresponding librations of the Moon. All these marginal zones are visible with greater distortions, caused bv perspective.

The chosen time for photographing allowed the AIS to obtain photos of a larger part of the lunar surface invisible from Earth and of a small part with already known formations (fig.153-155). Turned to the station was the lunar disk almost fully illuminated by the Sun. Under such conditions of illuminating the lunar surface its formations do not produce shadows and some details appears to be less noticeable.

The presence on photos of a part of lunar area visible from the Earth made it possible to tie down, never before visible objects situated on the reverse side of Moon, to already known ones and thus determine their selenographic coordinates. On the photo the boundary between visible and invisible (from the Earth) part of the Moon is designated by dotted line. Among the objects photographed from board AIS and visible from the Earth we have the Humboldt Sea , Sea (Mare) of Crises, Marginal Sea, Smith Sea, part of the southern Sea etc.

These seas, situated along the very edge of the Moon, are still visible during ground observations, and appear to us as result of perspective distortion as narrow and long, and their true form has not been determined until now. On photos, obtained from board the AIS, these seas are situated far from the visible edge of the Moon and their form is slightly distorted by perspective.

As results of preliminary investigation of available photos it can be mentioned, that on the invisible part of the lunar surface are predominant mountainous regions, while seas, similar to seas of the visible part, there are very few. Vividly expressed are crater seas, lying in the southern and near-equatorial regions.

Of the seas, situated close to the edge of the visible part, on the photos are clearly distinguished almost without distortions the Humboldt sea, Marginal Sea, Smith's Sea and Southern Sea. It was found that the southern Sea in a considerable part is situated on the reverse side of the Moon, whereby its boundaries have an irregular winding form.

Smith's Sea in comparison with the Southern Sea has a much rounder shape with a mountainous range cutting in into its southern tip. Smith's Sea in a large part spreads also on the reverse side of the Moon. The Marginal Sea has an elongated form with a depression opposite the Sea of Crises. Just as the Smith Sea it is

extended on the reverse side of the Moon. The Humboldt Sea has the peculiar shape of a pear.

The entire region adjoining the western fringes of the reverse side of the Moon, has a reflectivity, intermediate between the mountainous regions and the seas. By its reflectivity it is analogous with the area of the Moon, situated between the craters Tycho, Petavius and the Nectar Sea.

To the south - south-east from the Humboldt Sea on the boundary of the mentioned region stretches a chain of mountains of total length of more than 2000 km, crossing the equator and extending into the southern hemisphere . Beyond the mountain range extends, evidently, the continental peak with increased reflectivity.

In the region situated between 20 and 30° northern latitude and 140 and 160° western longitude is situated the Crater Sea with a diameter of about 300 km. In the southern end this sea is ended with a gulf. In the southern hemisphere, in the region with coordinates latitude - 30° and longitude + 130°, is situated a larger crater with a diameter of more than 100 km with dark bottom and bright central knoll surrounded by a luminous wide terrace.

To the east from above mentioned range, in the region of + 30° northern latitude, is situated a group of four craters of medium dimension; the largest one of that group has a diameter of about 70 km. To the south-west from that group, in the region with coordinates latitude + 10° and longitude + 110°, there is a separately situated round crater. In the southern hemisphere along the western edge are two regions with sharply reduced reflectivity.

Furthermore, the photos show individual regions with slightly increased and reduced reflectivity and numerous small details.

Fig.156 shows a chart of the reverse side of the Moon.

Investigating the operation of solar batteries.

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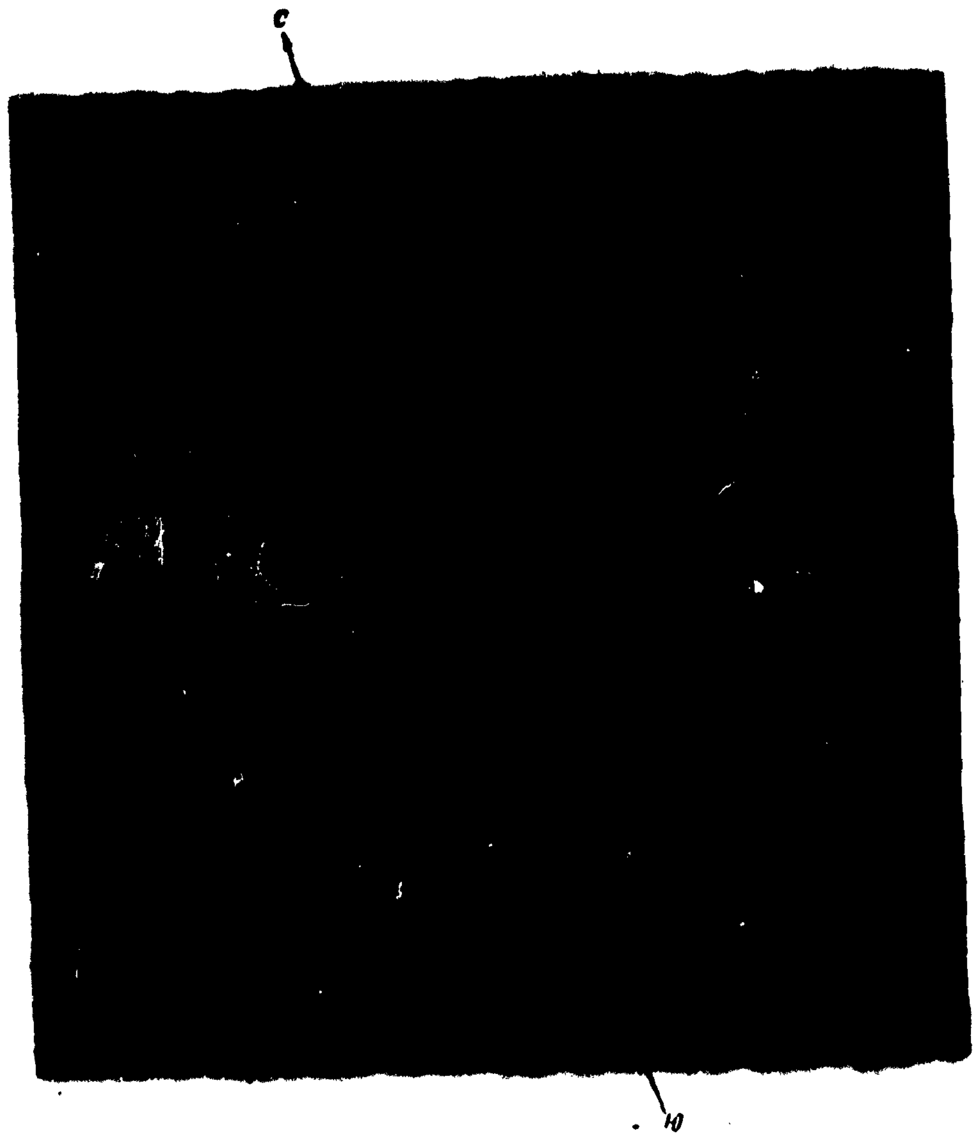

Fig. 155. Distribution of objects on the side of the Moon invisible from Earth:

1-larger crater sea diameter 300 km - Moscow Sea; 2-Gulf of Astronauts in Moscow Sea; 3- continuation of Southern Sea on reverse side of Moon; 4- Tsiolkovskiy crater; 5- Lomonosov crater; 6- Joliot-Curie crater; 7- mountain ridge Sovetskiy; 8- Dream (Mechta) Sea. Unbroken line - lunar equator, dotted line - boundary of visible from the Earth sections of the Moon; Roman numbers designate objects of visible part of the Moon: I - Humboldt Sea; II-Sea of Crises; III - Marginal Sea; IV - Sea of Waves; V - Smith Sea; VI - Fertility Sea; VII - Southern Sea.

As was stated before, on the third Soviet satellite and automatic interplanetary station for supplying power to the equipment, together with chemical current sources, were also used semiconductor (silicon) solar batteries. The installation of same appeared to be the first effort of employing solar batteries on man-made satellites and cosmic rockets under conditions of cosmic space.



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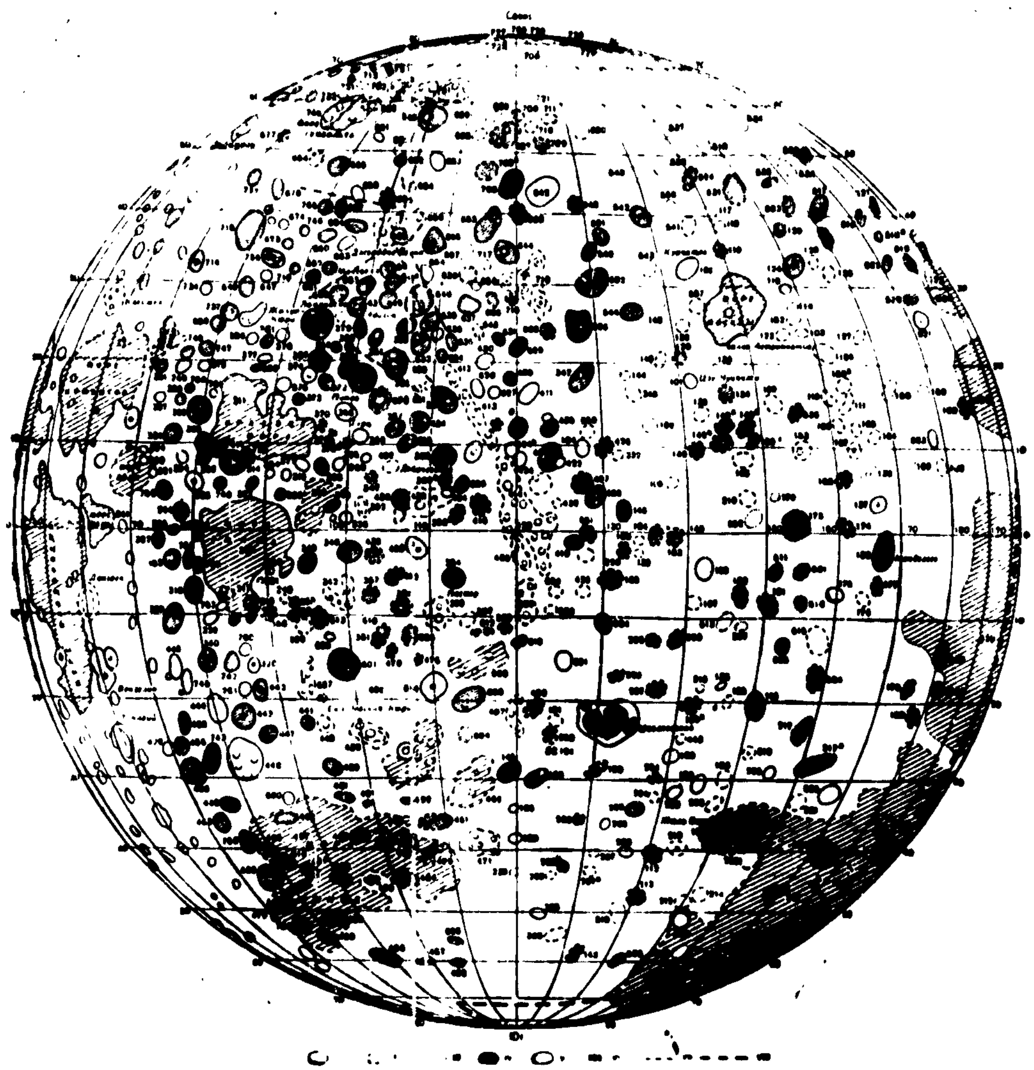

Fig.156. Chart showing the reverse side of the Moon

The problem concerning the possibility of using solar batteries under conditions of cosmic flight has a number of vague moments, which can be solved only by conducting direct experiments. Such moments first of all include the temperature condition of the solar battery, accurate calculation of which is extremely difficult, as well as the effect of meteoric erosion on the performance of silicon converter.

Operational data of sensing elements, mounted on the third man-made satellite of the Earth and on the AIS, enabled to obtain information about the temperature of solar battery. The average temperature of silicon converters varied between 16 and 30° C. Taking into consideration the very small thermal contact between



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sensing element and body , it is possible on the basis of already available data to confirm, that at a properly executed construction there should be no fear of failure of photo converters as result of overheating.

Measurement data, allowing to evaluate meteoric erosion, attest to the fact that the rubbing off of coatings, protecting the surface of solar batteries, is a slow process and also cannot be the cause for failure of the batteries. Preliminary conclusions can also be made regarding the effect of cosmic radiation. The operation of radio transmitter "MAYAK " through a period of many months confirms, that cosmic radiations, most likely, do not present greater danger for solar batteries.

The uninterrupted operation of "MAYAK" allows to make an important conclusion of the fact, that on Earth satellites it is already now suitable to use solar batteries of greater capacitances. Experience has shown, that such batteries are suitable not only for oriented, but in many cases also for non-oriented Earth satellites.

Positive experimental results on direct conversion of solar energy into electrical beyond the terrestrial atmosphere , experiments carried out on larger scale on the third Soviet Earth satellite and on the AIS, are of exclusive importance in solving problems of securing the work of scientific and measuring devices of satellites and cosmic rockets for a period of practically unlimited time interval.

## Chapter V. First Flight to the Venus

On February 12, 1961 the USSR launched an AIS toward the planet Venus.

The weight of the AIS exactly 643.5 kg. The lifting of same into interplanetary trajectory was realized with the aid of a guided cosmic rocket, which took off from the heavy man-made Earth satellite.

As was found from measurements carried out after the blast off, the trajectory of motion of the station was close to the calculated one. Moving along that trajectory, the AIS reached the region of the Venus in the second of the month of May 1961. Minimum distance of station from the Venus was less than 100 thousand km having covered a distance of 270 million km, which proves the high accuracy of guiding it along the trajectory.

By launching the AIS toward the planet Venus was established the first interplanetary road.

Equipment of AIS. The automatic interplanetary station (AIS), represents a cosmic apparatus, provided with a complex of radio technical and scientific devices, orientation and control system, programming instruments, temperature control system, electric power sources (fig. 157)

Structurally the AIS was made in form of an airtight body, consisting of a cylindrical part with two bottoms. In the body on an instrument frame was mounted the equipment and chemical battery units. On the outside of the body were situated part of the sensing elements of the scientific apparatus, two panels with solar batteries, louvers of the temperature control system and elements of the orientation system.

To one of the solar battery panels is fastened the block of thermal feelers to study the changes in optical coefficients of various coatings under conditions of longer stay in interplanetary space. On the outside of station body are also mounted four antennas. One of these - pencil beam antenna - has the shape of a paraboloid with a diameter of about 2 m and in intended for communication with interplanetary station at greater distances from the Earth and transmission of greater volume of information within a short time interval.

Two cruciform antennas, mounted on the solar battery panel, have a small radiation pattern and are intended for communication at medium distances from the Earth.

Omnidirectional antenna - 2.4 m long rod - intended for transmission of information and determining trajectory parameters at the section near the Earth.

Maximum dimensions of the station (without consideration of antennas and solar batteries) in length - 2035 mm and in diameter - 1050 mm.

The solar battery panel and rod antenna prior to break away of station from the cosmic rocket are folded up and open immediately after its separation.

The construction of the station secures the maintenance in the hermetic (airtight) body of initial gas pressure of about 900 mm Hg for the time of the entire flight.

The louvers of the temperature control system, mounted on the cylindrical part of the body, turn, open and close the radiation surface, correspondingly raising or lowering the transfer of heat, liberated during the operation of the equipment carried on board the AIS. The operation of louvers and fans, installed in the interior of the body, is controlled with the aid of an autonomous programming arrangement and system of temperature feelers, situated at points, subjected to maximum overheating or supercooling. In this way is solved the problem of providing normal temperature for the station equipment over the entire flight trajectory, during its trip from the Earth to the Venus, when the station comes close to the Sun at a distance of 110 million km, i.e. when the power of solar radiation rises by more than double.

Two solar battery panels, constantly oriented toward the Sun, secure continuous charging of the chemical current sources.

The radiotechnical installation of the AIS solves the following problems:

measures the movement parameters of the station relative to the Earth;

transmits to Earth measurement results, of measurements carried out by the

scientific instruments carried on board the station;

transmitting to Earth information about the operation of board instruments, pressures and temperatures within the station and on its body;

reception of Earth radio commands pertaining to control of the operation of the equipment on board the station.

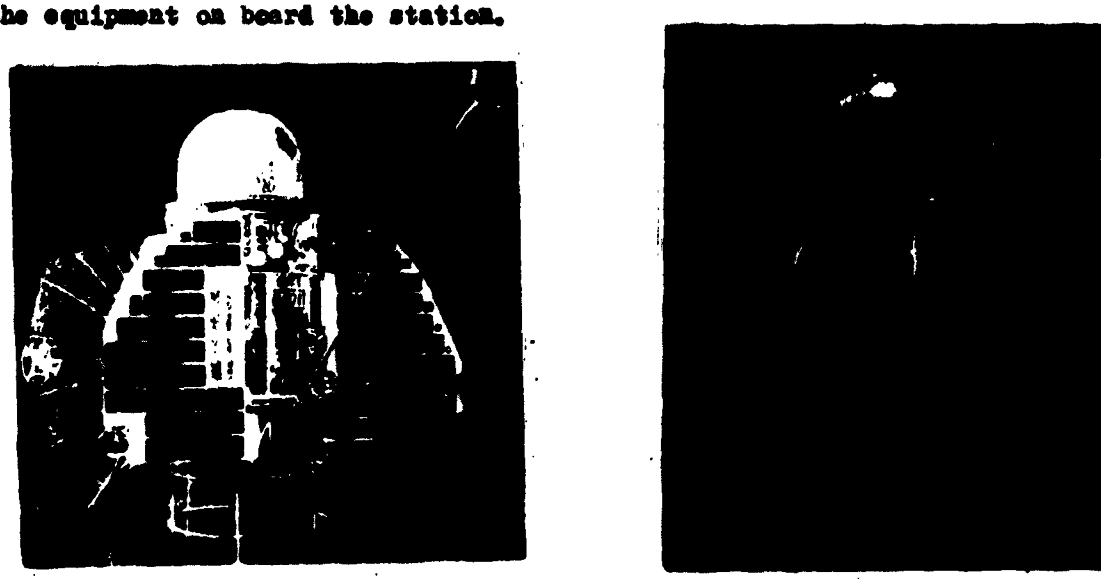

Fig.157. Automatic interplanetary station on assembly stand

top - forward view; below - rear view.

The operation of the equipment carried on board the station was controlled by the transmission of commands over the radio line from ground points, and by autonomous programming devices on board the station.

The orientation system of the AIS solves during flight along the trajectory the following problems:

elimination of any arbitrary rotation of the station, obtained during break away from the rocket, blasted off from a heavy man-made Earth satellite;

secures the seeking of the Sun from any position of the station and realizing station stabilization;

securing near the Venus orientation of the pencil beam (parabolic) antenna



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facing the Earth for the obtainment of much higher rate of transmission of scientific information and data about the operation of the airborne equipment back to Earth.

The AIS is equipped with a complex of scientific devices for carrying out physical measurements in cosmic space:

to study cosmic rays;

to measure magnetic fields in the range of several gamma units to several tens of gammas;

to measure charged particles of interplanetary gas and corpuscular streams of the Sun;

to register micrometeors.

On board the AIS is placed a banner with state emblem of the USSR. The banner represents a model of the Earth and is made structurally in form of a hollow sphere with a diameter of 7- mm made of titanium alloy. On the outer surface of the sphere is plotted an image of continental outlines. The surfaces of seas and oceans are painted in blue color, and the surface of continents - golden-yellow.

Within the spherical banner is placed a monumental medal with an image of the state emblem of the USSR. On the reverse side of the medal in the center is depicted a plan of the solar system with orbits of Mercury, Venus, Earth and Mars, and the inscription along the edges " USSR 1961".

The mutual disposition of planets correspond to the moment when the AIS draws closer to the Venus.

The spherical banner is placed in a special protective envelope, the outer surface of which is formed by pentagonal elements of stainless steel with image of USSR state emblem and inscription "Earth - Venus, 1961 ".

## Flight of Interplanetary Station Toward Venus

To carry out the flight to Venus it was necessary to select the flight trajectory, staisfying a number of conditions. If the date of rocket take-off and the date the AIS will aproach the Venus are mentioned, then the orbit of the AIS in the solar system, beyong the sphere of action of the Earth, is determined unilaterally. The AIS, getting away from the terrestrial sphere of action , should acquire a velocity, fully determined in magnitude, and in direction as well. The blast off and approach dates are selected so that the necessary escape velocity of the AIS from the terrestrial sphere of action would be possibly lower. The velocity magnitude, which the carrier-rocket should impart to the AIS along the acceleration stretch, will also be at minimum.

Of great importance is the method of accelerating the AIS by the carrier-rocket. At uninterrupted operation of all the rocket stages the weight of the useful load depends not only upon the velocity magnitude , which the AIS must acquire at the end of the acceleration section , but also upon the angle of inclination of the velocity relative to the horizon. At greater angles of inclination the terrestrial force of gravitation hinders acceleration, in connection with which it is possible that the weight of the useful load of the rocket decreases. So that the AIS should approach the sphere of action of the Earth, having a velocity in necessary direction, during continuous acceleration it may become necessary at the end of the acceleration section to acquire a velocity steeply inclined toward the horizon.

This can be avoided, if the method of accelerating and intermediate orbiting of the satellite, is applied. The satellite, carrying a cosmic rocket on board, is lifted by a carrier-rocket into circular orbit with minimum losses. Acceleration of the cosmic rocket, taking off from board the satellite, is done

in almost horizontal direction. Having properly selected the orbital plane of the satellite, place and time of blasting off the rocket from the satellite, it is possible to secure the escape of the AIS from the sphere of action in necessary direction.

Take off from board the satellite can be best realized by launching cosmic devices not only to the Venus, but also along the most variegated cosmic paths.

As already stated , the dates of blast off and approach to Venus are selected so, that the velocity magnitude of AIS's escape from the terrestrial sphere of action was possibly lower. This determines a number of ranges of take-off and approach dates, suitable from the viewpoint of rocket power. The acceptable intervals of take-off dates constitute 1 - 2 months and are repeated periodically approximately every 19 months. Of of such intervals is due at the end of 1960 - beginning of 1961. This interval was used for the launching of February 12.

From the terrestrial sphere of action the AIS escapes on an elliptical orbit of periodical motion around the Sun. For various energetically suitable trajecteries the time of flight up to approaching the Venus can be quite different. There is a flight trajectory over which the encounter between AIS and the Venus occurs during the first half of AIS rotation around the Sun, during the second half of rotation and so on.

For the blast off of February 12 was selected a trajectory, at which the encounter takes place during the first half of the rotation. At different kinds of trajecteries the flying time increases considerably and the deviations of the AIS at Venus do increase, all this depends upon the errors at the end of the acceleration section. In addition ,the distance from Earth to Venus at the moment the AIS approaches Venus for these trajectories, as a rule, is considerably greater, than the trajectory with encounter during the first half of the turn.

To secure the passing of the AIS in immediate vicinity of the planet, the lifting of the AIS into trajectory must be realized with greater accuracy. Errors in the velocity magnitude by 1 - 3 m/sec and errors in heading of velocity by 0.1 - 0.3° result in change in minimum distance between AIS and Venus by 100000 km. Such a deviation magnitude is also produced by a one minutes error in the time of rocket starting.

Deviations in the trajectory of the AIS from the Venus may also take place as result of the fact that the position of Venus is known only with specific accuracy. The basic source of these errors is the insufficient, for the given purpose, accuracy of measuring the astronomical unit (mean distance from Earth to Sun), determining the scale of the solar system.

At sufficiently accurate AIS trajectory measurements over a larger part of the flight the astronomical unit can be defined more closely.

For the launching of an AIS to Venus with the aid of a multistage rocket was first orbited a heavy man-made Earth satellite. The satellite travelled in orbit, close to circular, - with minimum distance from center of the Earth of 6601 km, maximum distance from center of the Earth of 6658 km and 65° inclination of the orbit toward the equator.

The cosmic rocket was blasted off from board the satellite in a precalculated point of the orbit. When the flight velocity of this rocket relative to the Earth became greater than the second cosmic velocity (escape velocity) by 661 m/sec the rocket escaped into a precalculated point in space, the power plant of the rocket was shut off, and from it broke away the AIS. Its free flight along a trajectory toward Venus has begun.

Thus was realized the first launching of a guided unit from board a man-made Earth satellite along an interplanetary path.

Further travel of the AIS is under the effect of gravitational forces of the Earth, Sun and planets.

Within the terrestrial sphere of action the AIS moved over a curve, close to a hyperbola, situated in plane , passing through the center of the Earth and unalternably oriented relative to the stars. This plane is close to the plane, in which the satellite moved.

As the AIS got away farher and farther its velocity with respect to Earth decreased gradually. The AIS reached the boundary of the terrestrial sphere of action on February 14, 2300 hrs Moscow time and had then a velocity of about 4 km/sec relative to the Earth.

The velocity of the AIS relative to the Sun , which is obtained by adding the velocity vector of the Earth relative to the Sun and the velocity vector of AIS relative to Earth, at the moment of escaping from the terrestrial sphere of action equalled 27.7km/sec

After this the movement of the AIS followed over an elliptical orbit with focus in the center of the Sun. This orbit has:

maximum distance from the Sun (distance in aphelion) - 151 million km

minimum distance from the Sun (distance in perihelion) - 106 million km.

inclination to the plane of the ecliptic (i.e. to the plane of the Earth's orbit - 0.5°.

Planes of motion of Earth, Venus and AIS we slightly inclined to each other.

On fig.158 is shown the movement of the AIS, Earth and Venus in projection on the plane of the Earth's orbit. The simultaneous positions of Earth, Venus and AIS are connected by straight lines. At the beginning of its movement around the Sun the rocket lagged behind the Earth. Not long from the spring equinox the Sun, AIS and the Earth were approximately on one straight line. The rocket then took over

the Earth in an angular movement around the Sun. The distance from Earth to AIS during its entire flight toward Venus rose continuously and at the moment of approach it reached 70 million km.

The angle between directions from the center of the Sun to Earth at the moment of take off and to the Venus at the moment of approaching it was 120°.The time of travel of the AIS up to the point of approaching Venus was not much over 3 months. The approach to Venus took place on 19-20 of the month of May 1961.

See page 174a for Figure 158

Fig.158. Movement of AIS relative to Sun (in projection on the orbital plane of the Earth)
1-position of Earth at moment AIS approached Venus; 2-position of Earth at moment of blast off; 3- line of AIS orbit angles; 4- position of Venus at moment of AIS approach; 5- position of Venus at the moment of blast off; 6- Sun; 7-orbit of Venus; 8-orbit of Earth; 9- direction into point of spring equinox.

Venus, just like the Earth, has a sphere of action (radius 600000 km). Within that sphere the influence of Venus on the Movement of AIS was predominat over the influence of the Sun.

Movement relative to Venus within its sphere of action followed along a trajectory, close to a hyperbola, with focus in the center of Venus.

Calculation of the orbit in accordance with obtained measurement data showed,

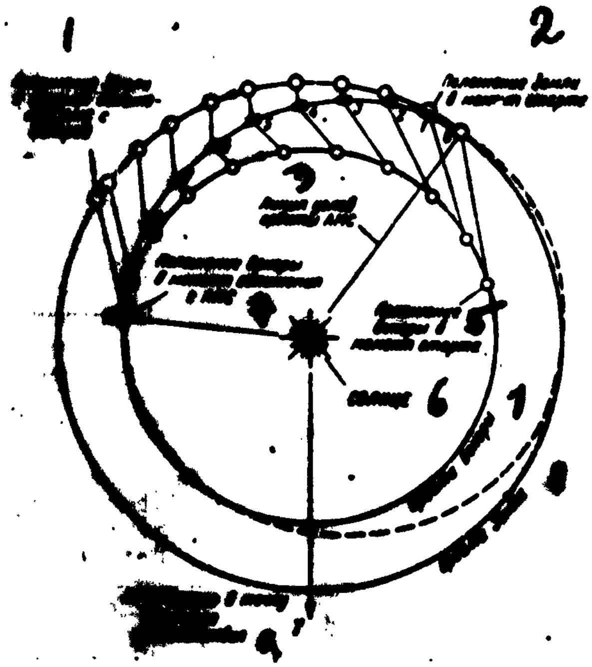


Fig. 158, Movement of AIS relative to Sun

that minimum distance from the AIS to Venus was less than 100000 km.

If the interplanetary station would be a bright point, then it would be possible from the ground to observe the movement of the station on the background of stationary stars. Its path over the celestial sphere (firmament) is shown on the astral chart (fig.159).

See page 175a for Figure 159

Fig.159. Visible movement of AIS (Unbroken line) and Venus (dotted line) over the celestial sphere (firmament). Numbers designate position of AIS and Venus within each 24 hours of flight. Along the vertical axis are indicated deviations ( in degrees). over the horizontal - direct ascent (in hrs.)

At the beginning of the movement the displacement of the station relative to stars was rapid. After leaving the terrestrial sphere of action the station was in the region of the firmament situated on the boundary of Cetus and Pisces constellations, in the center of a triangle, made up of beta Aries, alpha Pegasus and beta Cetus stars. By this time the angular displacements of the AIS over the firmament were already much slower. Along this section the AIS travelled relative to the Earth approximately along a radius.

Next the movement of the AIS over the celestial sphere, as . evident from the chart, became similar to the movement of planets. To the beginning of April the AIS was in the Pisces constellation, moving at so-called retrograde motion. The point

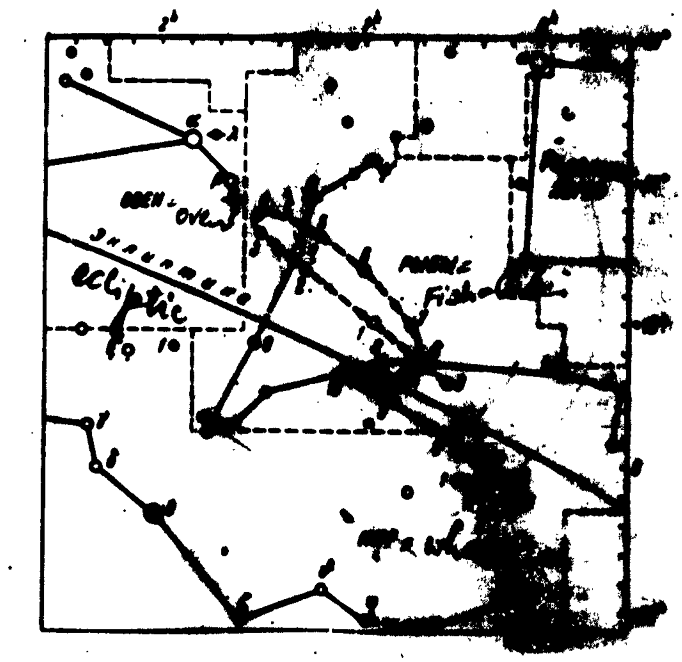

Figure 159 - Visible movement of AIS and Venus over the celestial sphere

where the retrograde motion changes into straight, bears the name of standing point. Direct movement among stars continued up to the point when the station approached Venus, which took place not far from the epsilon Pisces star.

Venus at the moment of AIS take off was in the Pisces constellation, moving among the constellations in a direct movement. The direct movement has slowed down gradually, and toward the End of March Venus came into standstill.

After the standstill began the retrograde movement of Venus, which lasted to the beginning of May 1961, then it changed into direct movement. On this section of direct movement of the Venus came about the approach with the AIS.

In table 35 are given rounded off values of distances between AIS and the Earth, Venus and the Sun and values of direct ascent angles within each 10 diurnal periods after the start.

Table 35. Distance of AIS from Earth, Venus and Sun

| No. of point on drawing. | Date (zero hrs universal time) | Distance of AIS from Earth, million km. | Dist. of AIS from Venus mill. km | Dist. of AIS from Sun mill. km | Dir. Ascent of AIS in hrs(h) & min (m) | Inclination of AIS |
|---|---|---|---|---|---|---|
| 1 | Febr. | 3,4 | 74 | 145 | $0^h27^m$ | −1,0° |
| 2 | Mar | 6,9 | 60 | 142 | $0^h22^m$ | −1,5° |
| 3 | Mar | 11 | 48 | 138 | $0^h16^m$ | −2,0° |
| 4 | Mar | 15 | 36 | 134 | $0^h10^m$ | −2,25° |
| 5 | Apr | 21 | 27 | 129 | $0^h08^m$ | −2,25° |
| 6 | Apr | 28 | 29 | 124 | $0^h10^m$ | −1,25° |
| 7 | Apr | 37 | 13 | 119 | $0^h18^m$ | 0,0° |
| 8 | May | 47 | 7,5 | 115 | $0^h32^m$ | 2,0° |
| 9 | May | 59 | 3,1 | 111 | $0^h51^m$ | 4,5° |
| 10 | May | 70 | less than 0,1 | 109 | $1^h09^m$ | 6,5° |

## Measuring-Control Outfitting of AIS

To control the AIS, establish its orbits and two-way communication with the AIS was constructed an automated radiotechnical metering arrangement.

The entire flight trajectory can be broken down conditionally into three

sections: section of flight of the heavy man-made Earth satellite; section where cosmic rocket starts from board the satellite and section through which the AIS moves under the effect of gravitational forces in direction to Venus.

Measuring the trajectory elements of the heavy man-made satellite was realized by special means , situated on the territory of the USSR. Data about the performance of units and components of the satellite were picked up by radio telemetering stations set up over the territory of our country, and by special ships stationed in the oceans.

The launching of the cosmic rocket from the heavy satellite was controlled by telemetering systems.

After separation of the AIS was activated a measuring set of the near ground section, intended for carrying out orbital and telemetering measurements. At each measuring point of the near ground section were set up radiotechnical transmitting and receiving-recording devices, parabolic antennas with program vectoring instruments.

Determination of actual trajectory when the AIS was removed from the Earth at a distance of 100000 km was realized by radiotechnical means of the Long Distance Cosmic Radio Communication Center. This center also picked up the telemetering information and controlled the equipment of the station. Over the command radio line were connected and disconnected the corresponding AIS instruments.

The operation of all media of the AIS was carried out in accordance with a special program, which determines the duration of the communication seances, their periodicity and operational conditions of the installation.

To pick up radio signal at greater distances narrow band low-noise receiver units were used. This calls for sufficiently accurate calculation of the values of received an emitted frequencies with consideration of the Doppler effect. To maintain constant frequency at the input of narrow band receiver filters, of receivers

situated on the interplanetary and at the metering point a prognosticating Doppler correction was introduced into the emitting and receiving frequency.

At points of the Long Range Cosmic Radio Communication Center larger antenna installations were constructed, allowing to pick up radio signals from sources, removed at huge distances from the Earth.

The antenna can be vectores into any given point of the celestial sphere with an accuracy of up to several angular minutes. Vectoring programs are automatically introduced into an electronic computer, which controls the antenna.

All measurement data are transmitted over an automatic line into a coordinating-computing center, where the trajectory measurements are processed with the aid of high speed electronic computers, forecasts are made regarding the movements of the AIS and antenna vectoring programs are calculated. The coordinating-computing center supervises all the ground metering services in accordance with a set up program.

## Chapter VI. Soviet Cosmic Ships-Satellites

### First Soviet ship-satellite

On May 15, 1960 the USSR sent up into orbit a man-made Earth satellite of the first cosmic ship. The launching was carried out for the purpose of finishing and testing the basic systems of a cosmic ship, guaranteeing its flight and return to Earth.

Total weight of ship-satellite after separation of last stage of carrier-rocket was 4540 kg. The ship satellite had a pressurized cabin with load, imitating the weight of a man, and equipment, necessary for man-flight into cosmic space. Weight of cabin (capsule) was 2.5 tons.

The ship-satellite was equipped with necessary board devices, total weight of which, together with power sources, was exactly 1477 kg. The equipment consisted of the following:

orientation system, securing specific position of ship during orbital flight;

braking power system, intended for decelerating the movement of the ship for the purpose of its change over at the given moment onto descending trajectory;

radiotechnical and radio electronic equipment, intended for measuring the orbit of the ship, controlling the operation of instruments carried on board ship, transmission to Earth of telemetering information and realizing communication with ship;

temperature and air conditioning control systems and many other systems.

Power for the equipment on board the ship was supplied from chemical current sources and from a solar battery, automatically oriented toward the Sun.

The radio transmitter "SIGNAL" mounted on the ship-satellite operated on a frequency of 19.995 mc in telegraph and telephone conditions.

The ship-satellite with the aid of a powerful carrier-rocket was lifted into given orbit, close to circular.

The initial value of orbital perigee altitude was 312 km, and the altitude of apogee - 369 km. The initial period of ships' round trip in orbit was 91.2 min, at a 65° orbital inclination.

After being lifted into orbit the ship-satellite was separated from the last stage of the carrier-rocket. The last stage moved in orbit, close to the orbit of the ship-satellite.

During the flight of the ship-satellite the ground measuring points, situated over the territory of the USSR, carried out systematic observations and reception of scientific information about the operation of the devices and equipment of the ship. After completion of the investigation program the ship-satellite was to begin its descent with separation of the pressurized cabin from it. No provisions were made for the return of the cabin back to earth. After studying the flight conditions in cosmic space, checking the functional reliability of the experimental cabin and separation of same from the ship-satellite, the cabin, as well as the ship-satellite, should cease their existence when entering the dense layers of the atmosphere along the descending trajectory.

In conformity with the flight program on May 19, at 2 hrs 15 min. Moscow time a command was sent up for the descent of the ship-satellite namely by connecting the braking power plant and separation of the pressurised cabin.

The decelerating power plant functioned normally. At the time of its operation was carried out the intended stabilization of the ship-satellite. Separation of the cabin from the ship took place at the specific moment of time. Normal operation of cabin stabilizing system has been recorded.

However as result of failure of one of the instruments of the orientation system which took place at that time, the direction of the braking pulse became deflected from the calculated one. In consequence instead of decelerating the ship there was a certain increase in velocity and the ship-satellite changes onto a new elliptical orbit, situated practically in the previous plane, but having a much higher apogee. The perigee of the orbit became equa 307 km, and the apogee - 690 km. The round trip period in orbit rose to 94.25 min.

The last stage of the carrier-rocket continued to move over the previous orbit. On July 17, 1960, during the 1019th round trip about the Earth it entered the dense layers of the atmosphere and ceased to exist.

Observation of the ship-satellite and reception of information from it after its transfer onto a new trajectory continued.

As result of the first launching of the ship-satellite were obtained important data:

tested was the take off and flight according to given program of a powerful carrier-rocket, which secured orbiting of cosmic ship with high accuracy;

In the process of flying was carried out reliable control of the ship-satellite and its orientation;

during the entire flight the air-conditioning and temperature control systems functioned normally;

the radio means of the ship-satellite, intended for transmission of commands to the ship, for controlling its orbit and transmission of telemetering

information carried out their mission successfully;

the auto-orientation of the solar battery on the cosmic ship was checked for the first time;

communication with ship-satellite in telegraph style was normal. In telephone style, when realising relay of ground station transmissions through the equipment of the ship-satellite, there was too much noise interference with greater distortions.

The launching of the first Soviet ship-satellite was the beginning of a greater and more complex operation on the creation of reliable cosmic ships, intended for man-flight.

## Second Soviet Ship-Satellite

On August 19, 1960 the USSR launched a second cosmic ship into oribt of Earth satellite.

The weight of the ship-satellite minus last stage of carrier-rocket was 4600 kg.

The ship was lifted into orbit, close to circular, with a perigee of 306 km and apogee of 339 km. The initial period of ship's rotation was 90.7 min., inclination of orbit to plane of equator - 64°57'.

The basic problem of launching the ship-satellite was further development of systems, guaranting the life activity of a person, as well the safety of his flight and return to Earth. During the flight were carried out numerous medical-biological experiments and realization of a program of scientific investigations of cosmic space.

To realize flight of a cosmic ship-satellite with living matter on board and safe return of same to Earth it was necessary to solve a series of complex scientific and technical problems, securing:

controlled flight of ship and its descent to Earth with greater accuracy at fixed point;

conditions for normal active life of living substances in cosmic flight;

reliable radio - and TV communication with cosmic ship.

All these problems have been solved successfully. Having completed the orbital flight, the cosmic ship together with its passengers - dogs Byelka and Strelka and other living substances have safely returned to Earth.

This historical achievement drew closer the time for direct conquering by man of the near solar space.

Faultless operation of all systems, guaranteeing the orbiting of the cosmic ship, as well as high structural data of a powerfull carrier-rocket enabled to reach an orbit, practically no different from the calculated.

## Arrangement of ship-satellite

The cosmic ship-satellite consisted of two basic components: cabin of ship and instrument section. In the cabin were situated:

apparatus securing active life of animals in flight;

equipment for biological experiments;

part of equipment for scientific investigations (photoemulsion units and radiometer);

part of equipment of orientation system;

devices for recording the characteristics of the cabin during descent (feelers of angular velocities, overloads, temperatures, noises etc.);

automatic systems, securing the landing of ship;

apparatus for autonomous registration of data regarding the functioning of instruments, as well as physiological data of the test animals along the descending section;

Ejection capsule with two dogs.

In the ejection capsule, in addition to two dogs, were 12 mice, insects, plants, mushroom cultures, seeds of corn, wheat, peas, onions, certain types of microbes and other biological objects.

Outside of the ejection capsule, in the cabin of ship, were place 28 laboratory mice and two white rats.

In the instrument section was arranged the radio telemetering equipment; apparatus controlling the flight of the ship; part of the equipment for scientific investigations (instruments for studying cosmic rays and short wave radiation of the Sun); temperature control devices; decelerating power plant.

On the outer surface of the ship were situated rudder nozzles and balloons (cylinders) with supply of compressed gas for the orientation system, sensing elements of scientific equipment, antennas of the radio system, experimental

solar batteries, as well as a thermo-insulation system to prevent burning up of the cabin along the descent section. In the walls of the cabin were situated heat resistant illuminators and rapidly opening airtight hatches.

The gaseous composition, humidity and air temperature in the cabin of the ship, necessary for normal vital activities of the experimental animals, have been provided by a regeneration and temperature control systems.

Transmission of information about the state of the experimental animals, the physical conditions in the cabin and in the instrument section, on the operation of the airborne equipment was realised with the aid of radiotelemetering systems to ground measuring points (measuring points on Earth). Radiotelemetering systems functioned in two ways:

a) direct transmission of telemetering information to the measuring points at the moment the ship flew over these points;

b) storing of information with subsequent reproduction and transmission of that information during flight of ship-satellite over the metering points.

The ship was equipped with the radio system "SIGNAL" intended for operational transmission of information and processing of radio telephone communication problems with satellites.

The transmit images of the experimental animals on board the ship a special TV apparatus was set up.

Ship control was automatic, and by the transmission of commands from Earth. On board the ship was installed a highly accurate-orbit control system.

Power for the equipment on board the ship was supplied from chemical current sources and from a solar battery. The solar battery was situated on two half-disks with a diameter of 1000 mm, oriented toward the Sun with the aid of a special system, regardless of the ship's position.

## Flight of Ship and its Return to Earth

After the ship is brought into orbit it separates itself from the last stage of the carrier-rocket. During orbital flight the task is being carried out in accordance with a fixed program of its basic systems; orientation system, telemetering system, temperature control system, scientific and TV apparatus, as well as the apparatus providing conditions for vital activities of the living organisms situated in the cabin of the ship.

Orientation of ship at the time of flight in orbit and along the descending section is accomplished with the aid of an orientation system. During the operation of the orientation system one axis of the ship was directed along the local vertical and the other one - perpendicularly to the plane of the orbit, the third one (longitudinal axis of ship) - perpendicular to the first two, along the intersection of the plane of local horizon and the plane of the orbit.

The flight of the ship-satellite was tracked by ground stations, situated over the territory of the USSR. The obtained information was transmitted automatically over communication lines into computation centers. As result of processing these data on electronic computers were obtained accurate elements of the ship's orbit, which provided the necessary prognosis of further movement of the ship in orbit and the possibility of its landing in the given region.

The requirements for exact knowledge on orbital elements depend upon the values of permissible errors during the landing of the ship-satellite, because to land in a given region it is necessary to select the exact time for cutting-in the braking (decelerating) power plant with consideration of the real values of coordinates and velocity of the ship-satellite at that particular moment of time. An error in ship velocity of 1 m/sec leads to a deviation of the

landing point by almost 50 km. An error in true altitude above the surface of the Earth, equalling 100 m, deflects the landing point by 4.5 km, and an error in direction of the velocity vector relative to the surface by the Earth by one angular minute leads to a deviation in landing point by 50-60 km.

In conformity with orbital prognosis data, as well as in conformity with telemetric measurements, which characterized the operation of the airborne equipment, from the coordinating-computer center in accordance with a preset program, control commands were transmitter to the ship-satellite in cosmic space.

During the 18-th round trip from the Earth was sent up a command for ship's descent with consideration of its landing at a given region (Fig. 160).

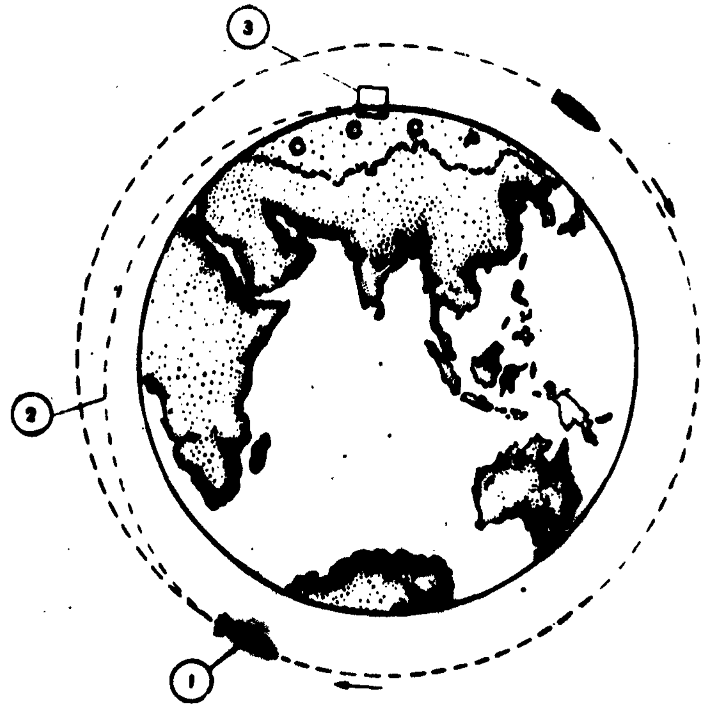


Fig.160. Flight and landing of cosmic ship-satellite:
1-deceleration with jet engine; 2-descending trajectory; 3-landing region for cabin of cosmic ship and the ejected capsule.

To make the ship-satellite descent from orbit to Earth with the aid of a braking power plant it was decelerated to a velocity of motion as required by calculation. The descending trajectory was selected so that the overloads during the entry of the descending ship into the dense layers of the atmosphere , and the time of their action should not exceed the values permissible for living organisms.

After the ship changes onto a descending trajectory the instrument section was separated from the cabin. The instrument section burned up when entering the dense layers of the atmosphere.

Along the descending section the cabin was decelerated in the atmosphere by a

special braking system. Dropping down to an altitude of 7000 m, the cabin flew from the moment descent began about 11000 km. Maximum overloads during cabin deceleration in the atmosphere amounted to 10 units.

At an altitude of 7-8 thousand km upon command from barometric relay stations the lid of the ejection hatch was thrown open and the capsule with the animals was catapulted out (ejected) from the cabin of the ship. The descending trip of the capsule was at a velocity of 608 m/sec, and that of ship cabin - 10 m/sec.

Immediately after the catapulting (ejection) of the capsule radio direction finding systems were cut in, intended for finding the direction of capsule and cabin at the time of descent and after their landing. Landing of animals, which made the flight on board the ship-satellite, could have been carried out directly in the cabin of the ship, but for the purpose of operating the catapulting system, which appears to be a reserve landing system for future man-flights, the capsule with the animals was catapulted (ejected) during flight.

The high landing accuracy of the ship-satellite (deviation of landing point from calculated was less than 10 km) indicates high perfection of the ship's control system and the accuracy of determining the orbital elements by ground metering units, the error of which have a direct effect on the deviation of the landing point. After the landing the cabin of the ship and the capsule with animals showed no signs of damage , which indicates the perfection of the landing system.

Providing conditions for vital activities on board the ship

For normal living functions of the animals specific atmospheric conditions in the cabin are needed. Therefore the basic requirements for an airtight ship cabin were as follows:

maintenance of barometric pressure, close to pressure at sea level, at an oxygen concentration of 20-25% and carbon dioxide concentration of not more than 1%;

maintenance of air temperature within limits of 15 - 25°C and relative humidity within limits of 30 - 70%;

purification of air from toxic admixtures, formed during the operation of cabin devices, as well as by the animals in the process of their life activities.

Two such dogs, like Belka and Strelka, require 8 - 9 liters of oxygen per hour and exhale during the respiration 6 - 7 liters of carbon dioxide per hour and 0.25 liter of water within a diurnal period. Taking into consideration that the normal life activity of a dog is disrupted upon a reduction in oxygen content to below 18% and at an increase in the content of carbon dioxide to 2-3%, it will become apparent, that without adoption of special measured in the cabin of cosmic ship the animals may perish rapidly.

To secure for the entire time of flight normal gas composition of the air, its temperature, pressure and humidity, the cabin was provided with an air conditioning system, which maintained the atmospheric parameters in the cabin within given limits.

To maintain required gas composition of the air in the airtight cabin of the ship it was provided with a special arrangement, in which highly active chemical compounds were used, absorbing the carbon dioxide and water vapors from the air of the cabin and generating an equivalent amount of oxygen.

The employment of chemical compounds for the regeneration of air in cabins of small volume encounters, however, considerable difficulties, one of which lies in the fact, that the rate of oxygen formation does not always meet the requirement of living organisms. To maintain equilibrium between the liberation of oxygen and the demand for same by animals it became necessary to create special devices, automatically controlling the rate of absorption of carbon dioxide and water vapors with the formation of the necessary amount of oxygen. This automatic control of the regeneration process is carried out by a very simple and reliable construction

of a sensitive element, reacting to change in operational condition of regeneration unit on the whole.

A reduction in the amount of oxygen and an increase in carbon dioxide concentration was absorbed (picked up) by a sensing element, transmitting corresponding signals to the telemetering and operational mechanisms. In case of excessive formation of oxygen the operational mechanism was automatically activated, as result of which the cabin was fed with air only partially enriched with oxygen.

The given air pressure in the cabin was maintained automatically. Especially developed dilters secured reliable purification of cabin air in case it became contaminated with toxic chemical impurities, liberated as result of the vital activities of the animals and during the operation of the devices.

Data about the operation characteristics of sensitive elements and about the parameters of the air in the cabin were transmitted over the telemetering system down to Earth.

Numerous experiments, carried out under laboratory conditions , showed, that the developed air conditioning and regeneration system secures reliable maintenance within given pressure limits, relative humidity, as well as oxygen and carbon dioxide concentrations in the air of the airtight cabin,

The problem of creating the necessary conditions in the cabin includes also the maintenance of given air temperature.

The dogs and other animals which took part in the flight were capable of enduring greater fluctuations of the surrounding temperature. However when readying for the flight the job was create maximum favorable temperature conditions. The fact is that considerable deviations of conditions from normal expose the animals to condition of more or less large additional load, requiring corresponding strain of the physiological mechanisms, controlling the vital activities of the organism.

This in turn, would create an unfavorable background for the endurance of basic conditions of cosmic flight - overloads, weightlessness etc.Consequently the problem came up of maintaining the given air temperature with very narrow fluctuations.

In solving this problem it was necessary to overcome a series of difficulties, a majority of which is connected with the inconstancy in the rate of heat liberation by the animal and the apparatus. But at the same time, in order that the air temperature should not go beyond the given limit, the amount of tapped (discharged) heat per each period should be within strict conformity with its entry.

To eliminate heat from the cabin of the ship was used a cooling unit with liquid-air radiator. The liquid refrigerant came to the radiater from the ship's thermoregulation system. Delivery of refrigerant was controlled in relation to the temperature in the cabin. Such a system secured stable air temperature maintenanse in the cabin during the entire flight.

To maintain given temperature in the instrument section and stable temperature of the coolant the ship was provided with a radiation heat exchanger and louver system.The heat from the airtight instrument section, filled with gas, was drawn off directly to the radiation heat exchanger, situated on the body of the instrument compartment.

Feeding and watering the experimental animals during the long flight on the man-made Earth satellite involve certain difficulties, connected mainly with the conditions of weightlessness. This eliminates the possibility of serving the dog water in an open vessel because the liquid can be carried away easily and it will become inaccessible for the animals. Solid food, intended for feeding under conditions of weightlessness, should not crumble and break into pieces.

A simple and effective method of surmounting the enymerated difficulties is the use of viscous, gel like mixture, containing the necessary nutritious sub-

stances in sufficient amount and simultaneously also the necessary amount of water. This combined method of feeding animals was applied for the first time to conduct a biological experiment on the man-made Earth satellite carrying the dog Layka.

On the basis of calculations and numerous experiments was developed a corresponding prescription for the combined feeding mixture. Such a feeding mixture has a jelly-like consistency and possesses sufficient cohesion with the walls of the feed box.

To portion out to the experimental animals their daily allotment of feed mixture the constructors developed a feeding automat.

To protect the feed mixture from spoiling it was subjected to sterilization in an autoclave at a temperature of 115°C, which offered reliable conservation of same.

When testing the animal feeding system under ground conditions it was found, that dogs, feed for a longer period of time on the combined mixture from the automatic feeding box, they lost no weight and suffered no thirst. It is necessary to point out, however, that the use of combined feed required and long systematic training of the animals in accordance with a special program under conditions , close to flying conditions on board a cosmic ship.

For the mice and rats were developed special cages. Along their walls were situated tubes - feed boxes, which were filled with dry nutrient briquettes, containing all the necessary nutritious substances. Water was in a separate little tank and flowed into the cage through a pipe with wick. The mice and rats were pre-trained to such a method of feed receiving.

Catapulting (ejection) capsule (container) for Animals

The catapulting container, in which dogs Byelka and Strelka were placed,

appears to be one of the container variants, developed for man-flight.

The shape of the container was chosen with consideration as to secure, after the catapulting, stable position of container axis relative to the velocity vector.

The container held the following units and systems:

cabin for animals with trough, feeding automat, sanitation arrangement, air conditioning system etc;

ejection and pyrotechnical means;

radio transmitters, intended for finding the direction of the container;

TV camera with illumination and mirror (reflecting) system;

blocks with nuclear photoemulsions.

Arrangement of the system is shown in fig.161. The cabin was made of sheet metal. In it were placed trough for placing the animals, feeding automat, sanitation arrangement.

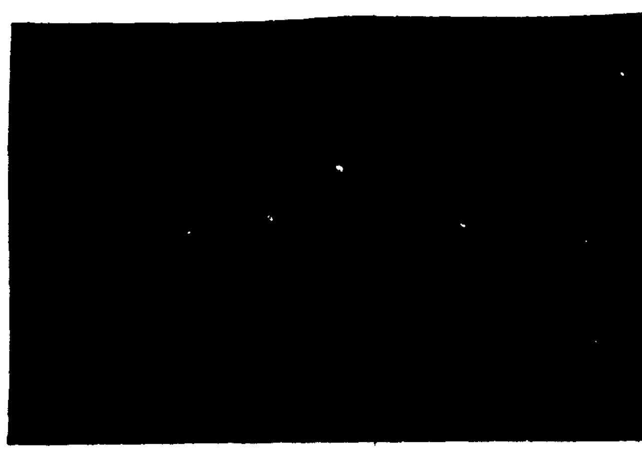

Fig.161, Hermetically sealed animal cabin in ejection container on board the ship-satellite:
1-bottle of air supply system; 2-the pyrotechnical mechanism of ejection; 3- radio direction finding unit; 4- special storage battery for heating microbe test tubes; 5-storage battery; 6- special scientific equipment unit; 7- ejection container; 8-movement feeler; 9-airtight animal cabin; 10-microphone; 11- radio direction finder antenna; 12-input and output valves; 13-TV camera; 14-mirror; 15-ventilation 16-combined feed mechanism.



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On the very trough were situated the movement feelers and automat for measuring blood pressure of the animals. On the upper bottom, made in form of a removable lid, were placed the TV camera, illumination and mirror system, fan and block of microorganism containers.

In the cabin were secured containers for small biological objects and a microphone, enabling to estimate the noise level in flight.

All the systems of the ejection container (fig.162) with animal cabin were designed for longer stay in cosmic flight.

Fig.162.Ejection container with animal cage
top - right view ; bottom - left view.

Television apparatus of cosmic ship

Objective data on the physiological functions of experimental animals cannot be fully generalized if no possibility exists for simultaneous direct observation of the animals. The TV system of the ship-satellite provided the physiologists with such an opportunity. Images, transmitted from board the ship at the time, when the ship-satellite was in the zone of action of ground receiving points, were recorded



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on motion picture film. Simultaneously on the very same film with an accuracy of up to 1 frame were recorded time markers, synchronized with the time markers, reproduced on telemetering tapes. In this way, by comparing the films it was possible to determine how the animal behaved at the given moment of time and what physiological changes accompanied these or any other activities of the animal.

When constructing the TV apparatus a number of contradicting requirements came up. On one hand, it was necessary to secure high quality of the image, on the other hand, to reduce to a maximum extent the weight, overall dimensions and, particularly, the power requirement of the apparatus. The scientific problem of transmitting information on the behavior of the animals and coordination of their movements, allowed for a considerable reduction in the parameters of the TV-image: number of scanning lines, frequency of frames thus sharply narrowing the spectrum of the TV-signal. Taking under consideration also the technical factors - in the first experiment it was found advisable to work in a possibly more narrow frequency spectrum, in order to guarantee against possible frequency-phase distortions, which could have originated during the transmission of spectrum of several mc.

The selection of such parameters provided the possibility of creating a highly economical and reliable radio channel with greater supply of energy with an image satisfying the requirements of the given problem.

On board the ship were placed two small size TV cameras. One, situated directly on the container hatch, through the window of the hatch were transmitted images of Byelka anfas = front view. The second camera was placed in the cabin of the ship and through the side window of the container transmitted images of Strelka in profile (fig.163) .

TV transmission began long before the take off of the ship. The condition of

the animals was observed on the take-off section, at the moment of change over from overloads into weightlessness and then during all the turns, when the ship-satellite was in communication with any one of the ground receiving stations (fig.164). Connection and disconnection of TV cameras and auxiliary illumination was realized upon commands from the Earth. The cameras were connected alternately.There was the possibility of switching over the cameras at any given moment of transmission.

At the ground stations, in addition to visual observation devices, were placed duplicated recording devices, in which all steps have been taken to secure highly reliable registration.

The obtained TV films are of greater scientific and perceptional importance, not to mention the impression, which the viewer experiences, having gained the possibility with "his own eyes" to take a look into the cosmos.

Great also is the purely technical value of the first experiment on the transmission from cosmos of images of moving objects. It yielded highly valuable experience , which will enable in the future to develop and improve cosmic TV systems.

### Medical-Biological Investigations

The basic problems of medical-biological experiment on the cosmic ship-satellite were:

studying the life activity characteristics of various animals and plants under conditions of cosmic flight.

studying the biological effect of basic cosmic flight facters on living organisms (overloads, lasting weightlessness, change over from reduced weightiness to increased and vice versa;

Studying the effect of cosmic radiations on animal and plant organisms (on the state of their vitality and heredity);

Fig.163. Pictures of Byelka and Strelka on TV screen

studying the effectiveness and functional characteristics of systems providing conditions for vital activities in flight (regeneration, temperature control, feedingm watering, sanitation systems etc).

To solve these problems the ejection container and hermetically sealed cabin of ship-satellite carried a number of biological species.

In the hermetically sealed cabin of ship-satellite were placed three cages holding two white laboratory rats, 15 black and 13 white laboratory mice. In the ejection container were housed: two dogs, cage with 6 black and 6 white laboratory mice, several hundred insects(prolific trush fly), two vessels with plant species - spiderwort, seeds of various types of onions, peas, wheat, corn and nigell ?, special vessels with actinomycetic fungi, single cell algae - chlorella in liquid and in solid nutritional media. 50 Cartridges held sealed ampoules with bacterial culture of intestinal bacilli (type KK-12, B, "aerogenesis", bacilli of oil-acid fermentation with staphylococcal culture, two types of phage (T02 and 13-21), solution of desoxyribonucleinic acid (DNK), as well as a culture of epithelial human tumorous



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cells (Hellcells) and small preserved sections of human skin and rabbit skin.

In addition, the ejection container contained four automatic bioelements with culture of oil-acid fermentation bacilli, with two bioelements situated in a special thermostat, and two in a nonheated container.

The experiment was preceded by greater preparatory work, including the development of investigation methods, control and recording equipment, as well as preliminary experiments, in which was investigated the effect of individual factors on the state of animals and plants, necessary background and control experiments.

When readying for the biological experiment on the cosmic ship-satellite in role of basic biological species were used traditional laboratory animals - dogs, the normal physiology of which has been well investigated.

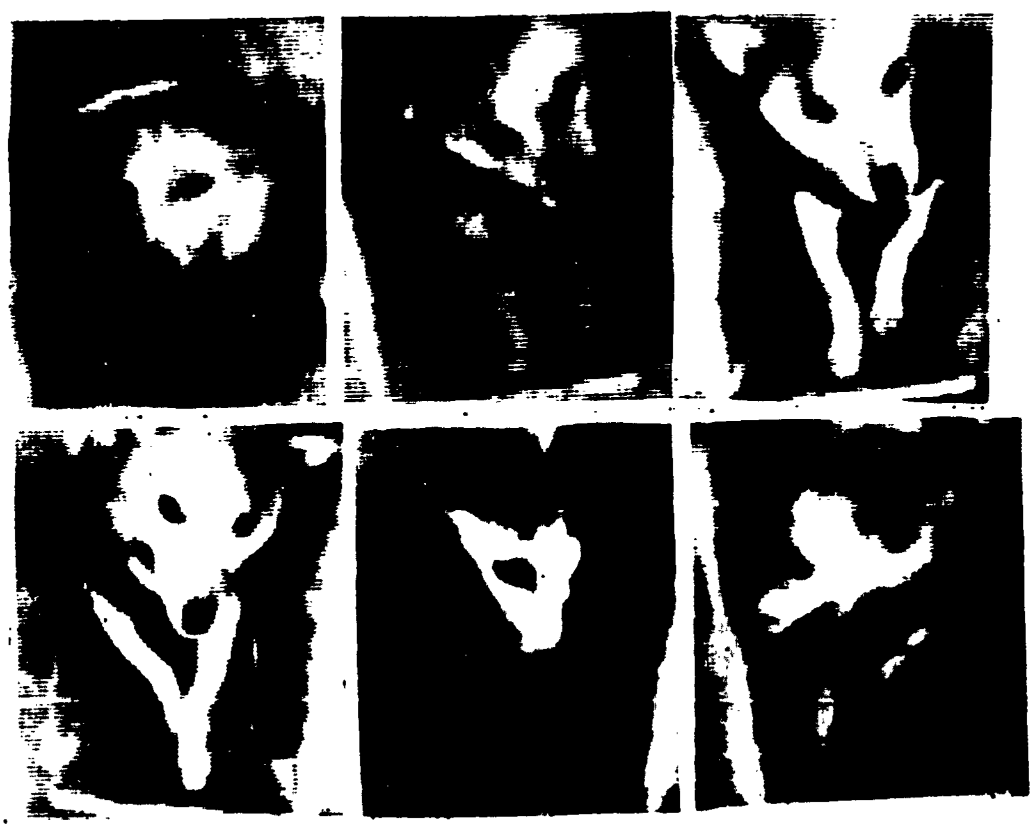

Fig.164.Byelka in various moments of flight



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These animals submit to training and resist various physical effects. The presently applied methods allow with sufficient accuracy and ease to record various physiological characteristics of dogs.

A whole series of requirements came up in connection with the experimental animals. The dimensions of the dogs had to provide a sufficient degree of freedom of movement in the cabin; color - qualitative and contrasting observation of the movements of the animals through the medium of TV. Preference was given to so-called " pure breed dogs " , which were distinguished by high resistance to actions of various outer conditions. Great importance was attached to the type of nervous activity: selected were dogs of strong, balanced, movable type in which the conditional reflexes necessary for the experiment have been easily developed.

For the experiment were taken mature dogs in the age bracket of from one and one half to three years. The animals were subjected to thorough physiological and clinico-veterinerial investigation. To record arterial pressure operations have been performed to bring out the carotid into the cutaneous part of the neck. For reliable registration of cardio biocurrents under the skin, were applied electrodes made of a special alloy.

As is known, during the flight on a cosmic ship the experimental animals should be confronted by a whole series of unusual factors: greater accelerations, vibrations, noise, long stay in a hermetically sealed cabin, feeding from automatic devices and realization of natural organismal functions in special clothing.

To train for the experiment the dogs underwent a longer period of training in a model of the cabin of the ship-satellite with a fixation system,allowing the animals to carry out the volume of movements necessary for normal life activity. The time of finding the dogs in fixed position was increased gradually.The dogs became accustomed to wearing sensing elements, fixing clothing and sanitation

devices . The program of animal training included also training of dogs in being fed with specially prepared mixtures from automatic devices, to which, as a rule, the dogs became easily and rapidly accustomed.

During the process of training was carried out a greater number of examinations to determine the resistance of dogs to accelerations. Each of the chosen animals was subjected several times to the effects of accelerations on a special stand. The experimental results allowed to establish the satisfactory endurance by experimental animals of overloads with slight individual fluctuations of the physiological parameters.

As is known, along the section of ship orbiting the organism of the animal is subjected to the effect of vibrations, which can definitely affect its condition. To explain this problem experiments were carried out by the results of which it was possible to estimate the satisfactory endurance of the animal of vibrations anticipated in flight. In addition, separate series of experiments have been made to examine the individual resistance of animals to the effects of shock overloads (taking place during ejection of the container), reduced barometric pressure, higher and lower temperature.

After completing the whole cycle of training and testing for participation in in the flight experiment were chosen dogs Byelka and Strelka (fig.165).

Both dogs passed preliminary training and testing jobs with satisfaction and were then placed in preflight condition.

To control the condition of the animals in flight and to solve the physiological problems of the experiment a special set of medical investigation devices was developed. This set secured the recording of the physiological functions of the experimental animals in flight of the cosmic ship.

During the flight were recorded the following physiological characteristics:

arterial pressure, electrocariogram, tonicity of the heart , respiratory frequency, body temperature, motorial activity of the animals.

Together with this were fixed data on barometric pressures, temperature and humidity in the airtight cabin, as well as control data on the functioning of systems providing conditions for life activities.

With consideration, that the basic purpose of animal experimentation is the training of man for flight into cosmic space, greater attention has been devoted to problems, connected with the study of the functioning of the motorial apparatus of the animals , and in particular the coordination of arbitrary movements. For this was used television and special movement sensing devices.

The TV films photographed on the ground allow to estimate the behavior of the animals in cosmic flight. In combination with informations received from movement sensing devices, they can provide material for judging about the state of higher functions of the central nervous system and about the adaptation of the animals to weightlessness conditions. Thanks to the presence on TV films of single time markers each movement of the animal can be connected with greater accuracy with the already available at the given moment values of any given physiological functions.

In the animal cabin in immediate vicinity of the dogs, as well as on the clothing of Byelka and Strelka we placed individual dosimeters for measuring ionizing radiation. Returned together with the animals back to Earth the dosimeters yielded data on the effect of charged particles on the animals, effect of electromagnetic radiation and of neutrons, included in the composition of cosmic radiation.

Investigation and evaluation of the biological effect of various factors, connected with cosmic flight, and above all the study of the biological effect of cosmic radiation, which represents a highly complex and varied problem, requiring the in-

volvement of the most variegated investigation methods: physical, general clinical, physiological, biochemical, microbiological, immunological, genetic etc.

Of great importance is the study of metabolism changes. It is important to explain, whether slight reversible functional changes do take place here or are there persitent metabolism displacements. For this purpose was selected a group of biochemical indices, which characterise the functions of the liver , endocrinal and nervous system which undergo considerable changes at greater loads against the organism, and under the effect of ionizing radiation as well.

Fig.165.First travellers into cosmos Byelka and Strelka after returning to Earth.

During a number of months prior to flight and when trained to endure the effects of individual flight factors (acceleration, vibration) the dogs were examined for the following characteristics: albumina fraction in blood serum, serum mucoid, cholinesterase activity of the blood, desoxycitidin in the urine.

A serious task was the examination of the state of the cardio-vascular system of the animals, which completed the cosmic flight. Prior to flight the animals were



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examined within a period of several months. The examination covered arterial and venal tonicity, vascular reaction and respond to compression, as well as skin temperature. After returning to Earth the dogs were again subjected to thorough examination of their cardio-vascular systems, and especially the state of the peripheral vessels. Examinations of Byelka and Strelka dogs after return to Earth revealed no noticeable changes.

A study of the immunological reactivity of the experimental dogs created the following important task. It was necessary to explain, whether the effects of cosmic radiation and other flight factors will bring about a depression of the natural non-susceptibility to microbes and development of infectious diseases as result of it. This is the more so important, since the future cosmonaut will have to remain for a longer period of time in confined quarters of a cosmic ship.

Strelka and Byelka were examined prior and after the flight to diagnose the phagocytic and bactericidid functions of the blood, and the bactericidic properties and natural microflora of the skin. These examinations on Earth were also carried out when the dogs were under the effect of accelerations and vibrations.

For all around study of various functional changes, occurring in the living organism at the time of flight, it is desired to gain data on a possibly larger number of animals. In these tasks in addition to the dogs, two white rats and mice were used.

The study of rats began several months prior to the flight. With the aid of a conditional reflex method was investigated the higher nervous activity of these animals, the tpological characteristics have been determined, blood analysis was made and an electrocardiogram was taken.

Already the first examinations after returning to Earth have shown, that the rats, as well as the dogs, have well endured the flight. During the flight they fed well

on the nutrients stored in the feed boxes . A thorough inspection of the rats revealed no scratches or contusions. The animals lost no weight, they were normally active.

The program of biological investigations on the second ship-satellite was also based on the use of microbiological and cytological investigation methods.These methods allow to solve effectively such important problems, as determination of specific time intervals living cells can remain in cosmic space, their growth and development under such conditions. They are also applicable in studying the genetic effect of cosmic space factors, especially the factor of cosmic radiations.

The characteristic of genetic effect of these radiations should be manifold, that is why, in addition to using animals ( e.g. rodents, insects etc), it is also possible to use microorganisms and living cells of human body in tissue culture. These and others possess certain advantages in connection with the greater rate of multiplication and corresponding rapid change in generations, Furthermore, a study of changes in the properties of microorganisms, especially such constant "satellites" of humans, as intestinal bacillus and staphylococci, is highly important in estimating their behavior in the organism of future cosmonauts.

In modern genetic investigations in the role of object special great attention is attracted by bacteriophages - ultramicroscopic agents, parasitizing on bacteria and entering with same into complex genetic relationships. Particularly sensitive indicators of genetic effect of radiation are the so-called lysogenic bacteria, which upon irradiation are capable of producing bacteriophages. Of familiar interest is also the study of the effect on growth and development of such living cells of accelerations, weightlessness, vibration etc.

In conformity with these deliberations on the second ship-satellite were placed

various microbiological and cytologival objects. They were especially prepared for this experiment, whereby the selection of these objects was guided by an effort to select organisms, widely used in laboratories of the whole world for the purpose of obtaining comparative results. In the number of objects were included intestinal bacilli cultures KK-12, for which the basic origin were, well known to microbiologists, bacteria with the most clearly expressed genetic characteristic.

This allows to make a quantitative determination of the degree of genetic changes and to compare these values with the level of radiation and quality of cosmic particles, registered on board the ship-satellite by physical instruments.

Through long and thorough examination of the returned cultures it will be possible to reveal the degree of changes in the number of so-called induced mutations, i.e. pathological in a majority of instances changes in hereditary properties. Furthermore, there is the possibility of investigating these cultures for the purpose of establishing the effect of radiation on the number of bacteriophages produced by them .

The varieties of intensitimal bacilli B and "aerogenesis", used in the experiment, also appear to be objects for studying mutation frequency.

To investigate the genetic changes in the most minute living substances -bacteriophages, the T-2 stamen was used.

Besides the T-2 was also used the bacteriophage stamen 13-21, specifically affecting the intestinal bacillus of the "aerogenesis" type . It was intended for studying the changes in the nature of lysis (dissolvement of bacteria, which takes place in presence of bacteriophages ).

This process for the phage 13-21 system - intestinal bacillus "aerogenesis" was documented for the first time by cytrapheric microphotographing and electron microscopy.

With respect to all the mentioned organisms was first obtained a detailed structural-physiological characteristic with the aid of the newest methods. Particularly, intestinal bacilli and staphylococci, which have also been exposed on the ship-satellite, were investigated under an electron microscope partially with the aid of the ultrathin microscopid section routine.

As to the oily-acid fermentation microbes used in the experiment, they were intended only for the development of automatic registration methods for the vital activities of microorganisms. The development of such methods offers the possibility of determining the life span of cells on long flying and non-returning satellites and rockets. The testing of oily-acid fermentation bacilli has been perfectly justified in this respect.

On that basis were developed an approved methods and special instruments - bioelements, which make it possible to record and transmit to Earth signals, characterizing the viability and physiological performances of the smallest living substances - bacteria for any length of time of rocket flight.

Bioelements after any exposure in flight can be activated by signals from Earth or by a programming device on board the ship.

On the second ship-satellite an effort was also made to use for genetic characterization of cosmic space living cells in tissue cultures. It is known, that heredity in such cells subjected to the effect of radiation changes a hundred times easier, than in microbes. But to preserve their vitality for longer periods of time without reseeding into new media is very difficult. To carry out such an experiment it was necessary to select well growing cells and suitable nutritious media for same. Taking into consideration this fact on the ship-satellite were used cancerous cells, conditionally called Hell's cells. These cells thrive well on artificial media and are widely used for studying genetic problems and for studying

the nature of cancer. For the cultivation of such cells was employed a method, enabling to obtain columns (accumulations) of cells on the wall of glass test tubes in which germination is carried out.

It was established during previous tests that the columns of cancer cells adhere to the walls of glass test tubes and ampoules with such a persistency, that they withstand vibrations, by much exceeding the ones, which take place during the blasting off of modern rockets. This offers the possibility during the processing of data to give a morphologo-biological characteristic of the cultures, a part of the development cycle of which took place in a specially constructed small thermostat on board the ship-satellite.

On board the ship-satellite were also exposed small sections of human and rabbit skin for the purpose of explaining the possible effect of cosmic space factors on the particularly sensitive cellular systems.

In our time biological, as well as genetic, investigations have been carried out in close cooperation with physico-chemical investigations. In particular, in the last decades it was shown that chemical substances can participate in the transfer of hereditary symptoms from one variety to another. Such a chemical substance is desoxyribonucleinic acid (DNK), included in the composition of nuclei of animal cells, plants and microbes. It is highly probable that this compound will first react to genetic effects of cosmic radiation. Taking this under consideration, on the ship-satellite were placed ampoules with desoxyribonucleinic acid, obtained from the goiter gland of a calf, with part of the ampoules filled with oxygen.

In this way, on board the ship-satellite was carried out a series of purposeful experiments on animals cells, microorganisms, bacteriophages and complex organic molecules in order to do everything possible for solving the problem of viability of cells and radiogenetic safety in cosmic space.

In addition to problems of explaining the effect of cosmic flight factors, first of all cosmic radiation, on the physiology of reganisms, foundations were laid for the study of the effect of these factors on heredity, and for solving the problem concerning genetic danger of cosmic flights.

Numerous investigations by Soviet and foreign scientists established, that such types of ionizing radiations , as x-rays, gamma rays, fast neutrons and certain other, represent a powerful source of hereditary changes in all organisms, including that of man.

Experiments in the bombardment of human tissues with x-rays showed, that a dosage of 10 Roentgens doubles the frequency in the origination of mutations. It was explained that various types of ionizing radiations have different biological effectiveness. For example, fast neutrons cause one and one half to two times more mutations than x- or gamma rays. The genetic effect of primary cosmic radiation could no be investigated until now. The flight of the second cosmic ship-satellite has, finally created the possibility for such an investigation.

In spite of the fact that a predominant number of mutations is harmful, some of these under specific conditions of the medium can be useful for viewing. Such useful mutations play an important role in the evolution of organic world and in the creation of new highly productive stamens of microorganisms and types of culture plants. Radioselection of microorganisms and plants in recent years becomes one of the tasks of selectioners. Therefore, in addition to explaining the genetic hazard of cosmic radiation, it is necessary to explain also the possibility of using same for radio-selection purposes.

On the ship-satellite were situated the following kinds of organisms, intended for first line genetic investigations: mice of two different lineages, small fruit flies-trushes also of two different lines, two tradescancia plants, wheat seed type

186, seeds of three types of peas, differing in radio stability, two types of corn - "Nemchinovsk" and "MOSCOW", batum and nigell onions, actinomycete fungi - producers of antibiotics.

What explains the selection of such objects for first genetic investigations, connected with cosmic flight ?

Mice and thrushes by virtue of a number of biological characteristics - high rate of multiplication and change in generations, easiness of their breeding and because of the enormous variety of their features, the heredity of which has been well investigated, are highly suitable for genetic investigations . Mice spending some time in the cosmos should be subjected to thorough cytological analysis for the purpose of explaining the changes, which might have taken place in cells of various tissues under the effect of cosmic rays. First of all it is necessary to make a thorough examination of the state of the chromosomic apparatus of blood producing organs.

As mentioned above, in the flight participated two kinds of thrushes (flies) One of these - line D-32 - is distinguished by very low mutability under natural conditions , the second - line D-18 - on the contrary, distinguished by very high natural mutability.

The tradescancia plant - classical object of cytological investigations ,since it possesses a small number of well distinguishable between each other chromosomes. In the animal cabin were especially placed plants with buds, because chromosomic transformations in tradescancia can be best observed in the cells being produced during the formation of pollen.

Dry seeds of culture plants - what, corn, peas - were used for the purpose of learning whether cosmic radiation causes any kind of changes (mutations) in various types and kinds of plants.

As to onions and nigellus, they are used basically for cytological investigations.

Ionizing radiation is widely used for the obtainment of new, more productive stamens of actinomycetes, offering such valuable antibiotics, as penicillin, streptomycin and others. On the cosmic ship were placed two stamens of fungi - producers of penicillin , highly distinguishable in radioactivity. Investigation of results of them being irradiated in cosmos will allow to solve the problem of biological effectiveness of cosmic radiations with respect to the given, very important object .

It should be pointed out, that each one of the enumerated genetic experiments is accompanied by strict control experiments with very same objects under normal conditions for same. This gave an objective evaluation of genetic investigation results.

Learning the laws of heredity and controlling same - one of the important problems of modern natural sciences. The entry of man into cosmos heralds the beginning of a new chapter in the development of genetics , a chapter, devoted to learning the laws governing the effect of cosmic flight factors on heredity and evolution, development of methods for protecting against the garmful effects of these factors and utilization of their positive effects. Genetic investigations on the second ship-satellite are only the first steps in that direction.

In the plan of long lasting future flights comes up the acute problem of generating air in hermetically sealed cabins and provide food for ship crew. Already simple calculations show, that the use for such purposes of chemical reagents and food supplies, taken on the Earth, would lead to a very high initial weight of the ship, because the reagents and food taken from the Earth will not be reproduced along the flight. In this connection within the scale of our entire planet such processes as carbon dioxide absorption, generation of oxygen and synthesis of complex organic substances from completely oxidized ones are realized in leaves of green plants as

result of photosynthesis .

Hence a hypothesis came into being that it is necessary to create on cosmic ships for purposes of regenerating air and derivation of food so called hot houses of green plants, which absorbing the carbon dioxide discharged by the living organism, would produce food and generate oxygen. Most suitable for such purposes are microscopic green algae, which develop very rapidly, are distinguished by greater photosynthesis activities and a number of other valuable qualities.

These considerations established the need for studying the effect of cosmic flight conditions on the preservation of vital functions of green algae. Chlorella situated on board the ship was placed in special ampoules in different physiological states: on diagonal agar and in a liquid nutritious medium and different suspension density. The algae were then exposed to light and darkness as well.

Scientific investigations on cosmic ship.

When studying cosmic rays a very important problem is the quantitative ratio of various nuclei groups in primary cosmic radiation.

Presently there are no exact data on the ratio of nuclear stream of nuclei belonging to the carbon, nitrogen, oxygen group to the stream of nuclei belonging to the lithium, beryllium, boron groups (most interesting from the viewpoint of origination of cosmic rays). Because of this it appears to be impossible to make a final conclusion about a specific mechanism of regeneration of nuclei during the movement of accelerated particles in interstellar space. To obtain new data in this field, it is necessary to know the magnitude of the ratio of streams of above mentioned nuclear groups with greater accuracy.

The second cosmic ship carried an apparatus , with the aid of which it is possible to obtain data on the composition of cosmic rays in the nuclear interval of

from helium to oxygen. For this purpose were used Cherenkov computers, controlled by a telescopid device made up of halogen gas discharge counters .

When cosmic radiation particles pass through the instrument in the given solid angle was activated a coincidence circuit, the pulse in which opened the channel of photo-multiplier. From the collector of photoOmultiplier was taken down the signal, originating when a nucleus flies through it. The amplitude of the pulse at the output of the Cherenkov counter is proportional to the square of the nuclear discharge. With the aid of a special device signals of various amplitudes have been converted into signals of corresponding duration, on which were superimposed signals from a standard generator. The number of pulses, filling up each one of the signals, was computed by a computing system and transmitter to the telemetering system.

Parallel with the measurements of the mentioned groups of nuclei were measured the streams of much heavier particles. With an integral Cherenkov computer were measured streams of nuclei with a charge of more than 5, 15 and 30.

Flight of the second cosmic ship and its return to Earth allowed to obtain in cosmic space photos of these processes, which took place in microcosm. For this purpose were used so-called nuclear photoemulsions. Flying through these emulsions, particles of cosmic rays experience collisions with atomic nuclei. As result of the collisions there is not only disintegration of the atomic nuclei but also new particles come into being. The originated particles also experience a number of conversions. In the emulsion take place new acts of reaction of particles, created as result of first collision, with the atomic nuclei of the substance.

With the aid of nuclear photoemulsions is possible to obtain sufficiently detailed photos of these phenomena. Studying the photoemulsions under a microscope it is possible to reproduce a picture of the processes, having taken place within billionth fractions of a second.

It is known, that the giant accelerators constructed on Earth, offer the possibility of obtaining particles with an energy below specific limit. In cosmic rays are encountered particles with an energy millions of times greater than that. The carrying of nuclear photoemulsions into cosmic space will allow for an effective use of this giant accelerator existing in nature.

On the second cosmic ship were placed several blocks of thicklayer photoemulsions, with direct development of photoemulsion in one of these on board the ship. Development of photoemulsion on board ship after a given time of exposure (of the magnitude of 10 hrs) allows more reliably to separate out traces of individual nuclei against a general background of cosmic radiation.

The autonomous programming device of the photoemulsion block after expiration of a given time issues a command, by which the piston situated in the interior of the cylinder separates the exposed layers and simultaneously introduces into the working volume the developer solution. The developing lasts 90 minutes, after which the developer is removed by the return movement of the piston. Then follows a command for secondary separation of layers and introduction of preserving solution. In the preserving solution the layers can be stored for several months, all the way to the beginning of final photolayer processing. During the processing can be investigated traces of relativistic nuclei of first cosmic radiation and data are obtained on the quantitative ratio of streams of various groups of nuclei.

In addition to the described photoemulsion block, on board the cosmic ship were also placed three more blocks, charged with a thick layer nuclear emulsion, which does not develop in flight.

The FE-2 block, intended for registration of elementary processes of nuclear reaction of high energy particles ( in the area of $10^{12}$ ev and over ), had an emulsion pile, composed of many layers of nuclear photoemulsion with a dimension

of 10 x 10 cm. The thickness of each layer was $400\mu$. Between the emulsion layers were placed thin, of magnitude of 1 mm, "targets" made of a light substance.

The presence in nuclear emulsion of silver and bromine atoms and the arranged "targets" offer the possibility of registering the case of reaction of high energy nucleons with heavy emulsion nuclei and with light "target" nuclei as well.

The high energy particles (neutral $\pi$-mesons) generated during acts of nuclear reaction give rise to photon showers, for the registration of which a special detecter was placed in the FB-2 block, this detecter was placed under the emulsion pile. This detecter consisted of seven lead plates 5 mm in thickness each (which corresponds to one cascade unit of length). Between the lead plates were situated nuclear emulsion and luminescent indicators of showers, thus facilitating the detection of concrete reaction acts.

Analysis of electron-photon shower instances, registered in nuclear emulsion, gives a certain quantitative characteristic of same, and about the energy, imparted during reactions to $\pi$-mesons. The knowledge of that energy, as well as an analysis of corresponding achievements, registered in the emulsion pile, offer the possibility of determining the individual parameters of the given nuclear reaction.

A comparison of obtained quantitative characteristics for acts of reaction of primary cosmic radiation particles with light and heavy nuclei will allow to explain the specificity and offer certain conclusions on the mechanism of this reaction.

Of special interest is the job of explaining the reaction nature of high energy multicharge particles , the study of which was impossible under ground conditions. To investigate multicharge particles in the composition of primary cosmic radiation the ship was equipped with F-1 and F-2 photoemulsion blocks. F-1 and F-2 blocks represented emulsion piles with a volume of 0.8 liters each.

One of the microphotos of a typical nuclear reaction, recorded in the emulsion, placed on board the cosmic ship-satellite, is shown in fig.166 .

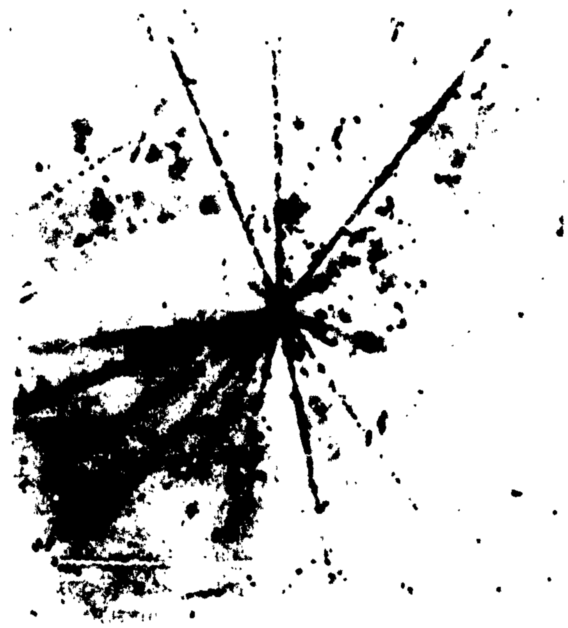

Fig.166. Microphoto of a typical nuclear reaction, recorded in emulsion, placed on board the cosmic ship-satellite.

The presence in interplanetary space of cosmic rays and radiation bands of the Earth may in many instances present a real danger for interplanetary space travellers. It has been proven experimentally in recent years, that sometimes originates a temporary increase in cosmic ray intensity, connected, most likely, with the development of solar activity. It was established, that at the moment of cosmic ray conflagration its intensity rises thousands of times.

Laws governing the time of cosmic ray conflagration cannot be established so far. But protection against solar flare ups of cosmic radiation is an absolutely real problem.

As is known, near the Earth exist radiation bands, representing zones of highly intensive radiation, consisting of charged particles, trapped in a trap, created by the terrestrial magnetic field.



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Investigations carried out on man-made satellites and cosmic rockets, show, that around the Earth there are two zones of high intensity radiation. The outer radiation zone by the radiation composition consists of electrons of wide energy spectrum. The stream of electrons in every direction constitutes $10^8$ - $10^{10}$ particles per 1 $cm^2$/sec.

Such stream of electrons is capable of creating a surface dosage of about $10^6$ roentgens per hour. But the electrons of the outer radiation zone are easily absorbed, and already under the protection of 1 g of light substance per 1 $cm^2$ of surface the radiation dosage in that zone will constitute only tenths of roentgens per hr.

Experiments, carried out on cosmic rockets, established, that the boundary and maximum radiation intensity in the outer zone change with time. This creates additional difficulties in considering the effect of radiation during cosmic flights. That is why one of the important problems is positive observation of the outer zone boundary and its radiation activity, particularly in the zone of high geomagnetic latitudes.

Particles, included in the composition of inner zone, preferably protons with energy of up to $10^8$ ev. Observed are also electrons, the energy of which does not exceed $10^6$ ev. Radiation in the inner zone is more rigid than in the outer. A dosage of radiation under protection of 1 g light substance per 1 $cm^2$ of surface constitutes here about 10 roentgens per hrs and decreases very slowly with increase in protection.

Protection against radiation in that zone requires the use of a greater number of substances. Long lasting flights in the inner zone without special protection are connected with considerable radiation hazard.

In this way, the instability of radiation band boundaries and accidental increases in cosmic radiation activity make cosmic radiation level control and a

detailed investigation of the lower boundaries of radiation bands a highly important and actual task.

To solve these problems the cosmic ship carried a dosimetering apparatus (radiometer). The components of the radiometer include two gas discharge and two scintillation counters. One of the gas discharge counters is placed under the additional absorbent (screen), consisting of brass and iron. The scintillation counter with photomultiplier and sodium iodite crystal were placed in one block with the gas-discharge counters. The second scintillation counter with photomultiplier and cesium iodide crystal with a thickness of 2 mm were situated on the outside. So that the counter should not be affected by visible light, the cesium iodide crystal was covered with an aluminum foil 7μ in thickness.

Gas discharge counters, as well as the scintillation counter with sodium iodide crystal yield information about the number of particles, which passed through them. At the same time the scintillation counters allows to estimate total ionization, caused by the passing particles.

The obtained information on the number of passed through particles and total ionization, caused by these particles in the crystals, will yield quantitative data on the level (dosage) of cosmic radiation.

Study of shortwave radiation is of great scientific and practical importance. In this zone of the spectrum is concentrated the basic radiation of the solar corona and chromosphere - very little investigated outer shells of the Sun.

This radiation causes certain important processes, takin place in the terrestrial atmosphere, especially the formation of ionosphere.

On board the cosmic ship were mounted two types of instruments for studying shortwave radiation of the Sun.

In the apparatus of first type the receiver of shortwave radiation was an

open type electronic multiplier with electrodes made of activated beryllium bronze. In front of the input of the electronic multiplier (electron multiplier tube) was placed a disk with a selection of various filters for the separation of corresponding zones of the shortwave solar radiation spectrum. With the aid of a relay-scanner mechanism the disk made a small angle turn each second, shifting a new filter in front of the electron multiplier tube. The following filters have been used in the apparatus:

copper foil 0.15 mm thick to separate the zone of the spectrum from 1.4 to 3 Å;

beryllium foil 0.06 mm thick to separate spectral zone shorter than 12 Å;

aluminum foil 0.005 mm thick to separate spectral zone from 8 to 20 Å;

polystyrene layer with thin carbon layer applied on it to separate spectral zones from 44 to 100 Å;

lithium fluoride plate 0.5 mm thick to separate lines $L_\alpha$ with wavelength of 1216 Å;

calcium fluoride plate 0.5 mm thick, which considerably weakens radiation with wavelength of 1216 Å passing through it and allows to evaluate the background in the region of line $L_\alpha$ and at the same time measure more accurately the radiation intensity of that line;

quartz crystal plate 0.5 mm thick, to separate radiations with wavelength of more than 1500 Å.

The last filter was designated mainly for the purpose of calculating the change in angle of incidence of radiation on the filter and receiver, connected with the rotation of the satellite in nonoriented state. The apparatus had six pickups set up at various points of the cosmic ship so that their fields of vision were not superimposed over each other This offered the possibility of increasing the prob-

ability of solar radiation falling on the pickups at any orientation of the cosmic ship in space. The sensitivity of the pickups was limited in the long wave zone of the spectrum, so as to reduce the background from longwave solar radiation. Signals from the pickups (receivers) went into a radiotechnical circuit, at the output of which was generated a voltage , proportional to the radiation intensity of the radiation falling on the photocathode. Measurement results were then transmitted to Earth by a telemetering system.

The outfit included a control unit which secured the connection of the corresponding pickup, mechanism of changing filters and other circuits only at the time, when they were illuminated by the Sun. In addition there were optical sensing elements to determine the radiation angle of incidence on the filters.

The apparatus of second type was designated for measuring the soft x-ray radiation intensity of the corona near the tip of the spectrum, preferably at the time of the flashes.In this apparatus were used the most sensitive radiation receivers - photon counters for the given zone of the spectrum, representing self-quenched Geiger counters with input windows of beryllium foil, serving as filter. Measurements were carries out in two spectral zones - 10 - 6 Å and 6 - 3 Å. To each of these spectral zones corresponded six counters, which were grouped (bunched) in three blocks, each one having two counters placed under right angle relative to each other, for the first and two counter each for the second zone of the spectrum.

When a photon penetrated into the counter and fell in the gas, filling up the counter, a short electric discharge originated. The obtained current pulses went into a radio block. In the radio block the signal became amplified and went into the computing system, consisting of trigger cells. This system computed the number of pulses, having passed during the time of exposure. A corresponding number in the double computation system was recorded on an autonomous memory device, which stored

all recordings for a period of 24 hours up to the moment of their transmission to Earth over the telemetering system. Exposure time was 180 sec, which warranted registration of x-ray radiations of the Sun with sufficient time resolution .

To protect the input windows of the counters against x-ray radiation, originating during the bombardment of these windows (as well as parts of the equipment surrounding same ) by fast electrons, existing in the radiation bands of the Earth, a system of magnets and diaphragms has been provided and placed in front of each counter. The magnets deflected aside all electrons with energy not exceeding 15 - 25 thousand ev. To register the background, produced by higher energy electrons, on the outer shell was placed an electron scintillation counter.

Data obtained with the aid of the described arrangement, data about changes in solar activities in the shortwave zone of the spectrum were compared with data of ground observations of the , ionosphere, visible by chromospheric flashes and other phenomena, connected with the activity of the Sun to reveal correlation between the processes, taking place in the outer shells of the Sun and in the terrestrial atmosphere.

## Further Launchings of Ships-Satellites in the USSR

After successful flight of the second Soviet ship-satellite in conformity with the plan of operations connected with the study of cosmic space the Soviets launched three more cosmic ships lifting same up into orbit of man-made Earth satellites. The basic mission of these launchings was further development of ship-satellite construction and development of its basic systems, securing flight and descent of the ship, with the purpose of readying for man's cosmic flight.

The launching of the third Soviet ship-satellite was executed on December 1, 1960. It weight minus the last stage of carrier rocket was exactly 4563 kg. Altitude or orbital perigee was 187.3 km, altitude of apogee - 265 km, period of rotation -

- 88.6 min.

To carry out medical-biological investigations under conditions of cosmic flight the cabin of the ship held experimental animals - dogs Pchelka and Mushka, as well as other animals, insects and plants. Observation of the animals during flight was carried out with the aid of a radio-television set up and telemetering system. The radio transmitter "SIGNAL" installed on board ship functioned on a frequency of 19.995 mc in telegraph style.

Around 1200 hrs Moscow time on December 2, 1960, the ship-satellite continued in its orbital journey.

By that time all the tests intended by the program of testing the construction of the ship and its airborne equipment, as well as the medical-biological investigations, have been fully completed. Additonal data have been obtained on structural and functional reliability of the individual units and systems of the ship in flight.

The results of processing the telemetering and TV-information, obtained from the ship-satellite, showed, that the dogs, just as during the previoud flight, have well endured the period of orbiting. The basic physiological characteristics,characterizing the condition of the experimental animals, during their many hours of stay under weightlessness conditions were close to ordinary values. The behavior of the animals was calm, and their movements - coordinated.

After obtaining the necessary data a command was issued for the descent of the ship-stallite back to Earth. In view of the fact that the descent took place along a non-calculated trajectory, the ship-satellite ceased its existence upon entry into the dense layers of the atmosphere.

Launching of the fourth ship-satellite was realized on March 9, 1961. Its weight was exactly 4700 kg. The ship was brought into orbit with a perigee altitude of 183.5 km and apogee altitude of 248.8 km. Inclination of orbit to the plane of

equator was 64°56'.

The ship-satellite carried an experimental animal in its cabin - dog Chernushka and other objects of biological experimentation.

The entire board equipment of the ship functioned well securing the fulfillment of the given flight program. The separation of the ship from the last stage of carrier-rocket, cut-in of orientation system, connection of automatic devices along the descending section were realized exactly at specific time periods. The temperature control system maintained constant temperature in the cabin - within limits of +(16 - 20)°C. Humidity of air on the cabin was 37-40%, and pressure 760-770 mm Hg. Chernushka felt normal during the entire duration of flight. Immediately after reaching orbit her pulse was 120, and respiration frequency 50 - 60.

After completing the investigations mentioned in the program the ship was brought down on the very same day and landed in a specific region.

On March 25, 1961 the fifth Soviet ship-satellite was hoisted into orbit. The weight of the ship was 4695 kg minus the weight of the last stage of the carrier-rocket. The orbital parameters were close to calculated: altitude of perigee 178.1 km, altitude of apogee 247 km, period of rotation 88.42 min.

The ship-atellite carried in its cabin the dog Zvezdochka and other biological objects. The antire board equipment of the ship functioned well during the flight. After completing the flight program the ship-satellite upon command from the Earth was brought back from orbit and landed at a designated point of the USSR.

During the launching of ships-satellites on March 9 and 25, 1961, the pilots seats were occupied by manikins. The flights were carried out in accordance with the very same program, which was earmarked for the first flight of a ship with a cosmonaut on board. Both flights, which followed strictly according to program,

gave proof of the high reliability of the ship and allowed to undertake steps toward the realization of man's first flight on the ship-satellite "VOSTOK".

## Chapter VII. Man's First Flight into Cosmic Space

On April 12, 1961, the USSR, for the firt time in the history, realize the flight of man into cosmic space. The cosmic ship "VOSTOK" with Soviet pilot-cosmonaut Yu.A. Gagarin on board was shot up into orbit of an Earth satellite. Weight of ship-satellite less the last stage of carrier-rocket was 4725 kg. Altitude of orbital perigee was, according to definite data, obtained on the basis of processing all the measurements, 181 km, altitude of apogee - 327 km, orbital inclination - 64° 57'.

The rocket took off from the BAYKONUR cosmodrome, situated in the region of 47° northern latitude and 65° eastern longitude. The carrier-rocket had six power plants with total output of 20 million hp.

Having completed its orbital flight, the ship-satellite landed safely in the region of the village Smelovka in the Ternovsk region of Saratov province.

The first cosmic flight of a Sovier citizen opened an era for man's direct penetration into cosmic space, and it appears to be a great achievement in the history of civilization. Realization of this flight is the result of a greater planned program of operations on the conquering of cosmic space, being conducted in the USSR.

Arrangement of cosmic ship VOSTOK

The cosmic ship VOSTOK has been constructed on the basis of experience, derived during the launchings of first Soviet ships-satellites.

The ship-satellite consists of two basic parts:

pilot cabin, in which the cosmonaut was situated, equipment to provide living conditions and landing syste ; instrument section, intended for the installation of instruments, functioning during orbital flight, and braking power plant of ship.

After reaching orbit the ship-satellite breaks away from the last stage of

carrier-rocket. During the flight the equipment on board the shipfunctions in accordance with a specific program, securing the measurement of orbital parameters, transmission to Earth of telemetering information and TV-images of the cosmonaut, two-way radio communication with Earth, maintaining given temperature on board ship, conditioning of air in pilot cabin. The performance of the equipment was automatically controlled , with the aid of program devices on board the ship and if necessary by the pilot-cosmonaut.

The program of man's first flight called for one round trip around the Earth. But the construction and equipment of the ship-satellite allow to carry out much longer flights.

After completing the flight program, prior to landing, a special system was employed for orienting the ship in a specific direction. Then at a given point in the orbit the braking power plant of the ship is cut in, thus decelerating the ship to a value set by calculation. As result of this the ship begins the downward trip along a descending trajectory (fig.167).

The cabin with cosmonaut is slowed down in the atmosphere. The descending trajectory is selected so, that the overloads during the entry of the apparatus into the dense layers of the atmosphere have not exceeded the overloads, permissible for humans. After the ships capsule drops to a given altitude the landing system is brought into operation. Direct landing of the pilot capsule takes place at a low rate of speed. From the moment of cutting-in the braking power plant to landing the ship covers about 8000 km. The duration of flight along the descending section is approximately 30 min.

The outer surface of pilot's capsule is coated with a thermal protection layer, which protects it against burning up during its movement along the descending section in the dense layers of the atmosphere.The capsule is provided with three

illuminators and two rapidly opening hatches. The illuminators are equipped with heat resistant glass and make it possible for the cosmonaut to conduct observations during the entire duration of the flight.

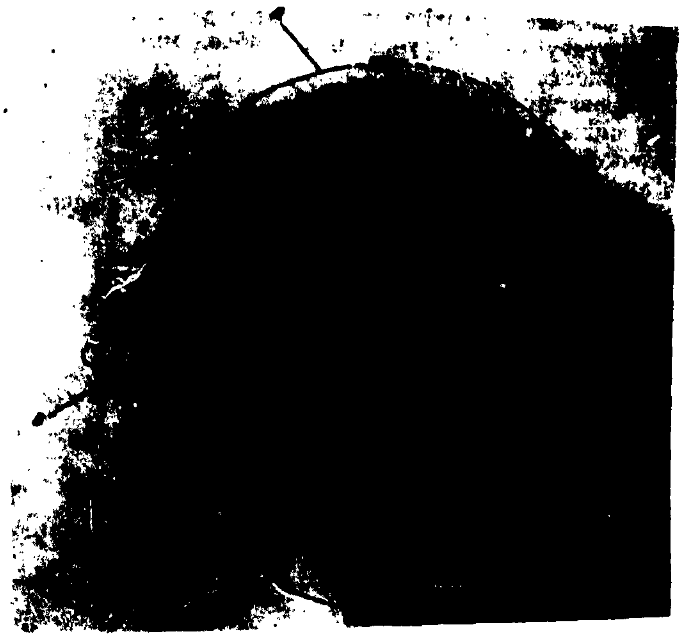

Fig.167.Schematic drawing of the flight of ship VOSTOK

A-orbiting section; B- connection of braking power plant; C-descent seat.

The cosmonaut occupies an ejection seat in the ship-satellite, and this seat appears to be his working place in flight, and if necessary it also serves for abandoning ship by the cosmonaut. The seat is arranged in such a way that the over loads during orbiting and descending affect the cosmonaut in most favorable direction (chest-spine).

During the first flight the pilot-cosmonaut wore a protective suit, securing the preservation of his life and activities, even in the case the cabin becomes depressurized during flight.

The ship-satellite carried also:

apparatus and equipment necessary for the viability of the human organism (air conditioning system, pressure control system, food and water, a system for removing discharge products );



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flight control apparatus and system for manual control of ship (pilot's instrument panel, instrument panel, manual control block etc.);

landing systems;

radio apparatus for Earth-cosmonaut communication;

system for autonomous registration of data on the functioning of instruments, radio telemetering systems and various sensing elements;

TV-system to observe the cosmonaut from the Earth;

apparatus recording the functions (physiological) of man;

braking power plant of ship;

orientation system apparatus;

flight control apparatus;

radio systems for measuring orbital parameters;

temperature control system;

electric power sources.

On the outer surface of the ship were installed control organs, elements of orientation system, louvers of temperature control system and radio antennas.

The pilot's cabin on board the ship-satellite is more spacious than the cockpit on an aircraft. The outfitting of the cabin is made with consideration of the operational convenience of the cosmonaut in flight (fig.168). Situated in his chair, the cosmonaut can execute all the necessary operations connected with observation, communication with Earth, controlling flight and in the case of necessity - to control the ship.

In the body of pilot seat are built in: separable back with belting (tying) system for fixing the body of the pilot during ejection and descent by parachute;

parachute system; ejection and pyrotechnical devices;

emergency (carried along) supply of food, water and equipment, and radio means

for communication and direction finding, which the cosmonaut can use when landing; air conditioning system for protective suit and parachute type oxygen unit; automatic mechanisms of the seat.

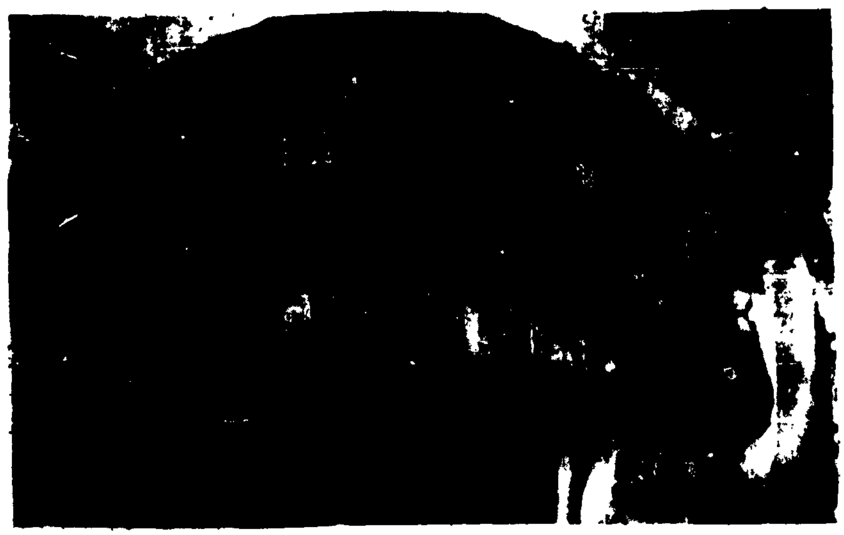

Fig.168. Internal view of cosmonaut cabin on board VOSTOK ship-satellite

1- pilot's panel; 2-instrument panel with globe; 3-TV-camera; 4-illuminator with optical orienting point; 5- handle for controlling the orientation of ship; 6-radio receiver; 7- containers with food.

Landing of cosmonaut may take place in the capsule of the ship. Such a landing method was tested on the fourth and fifth Soviet ships-satellites, the cabins of which housed experimental animals. Provided is also a landing variant by ejection of seat with cosmonaut from the cabin at an altitude of about 7 km and subsequent parachute descent. This variant was also checked during the launchings of ship-satellites.

The air-conditioning system, installed on the ship-satellite, maintains in the pilot's cabin a normal pressure, normal oxygen concentration at a carbon dioxide



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concentration of not more than 1%, temperature at a level of 15-22°C and relative humidity within limits of 30-70%. Regeneration of air composition - absorption of carbon dioxide and water vapors with generation of corresponding amount of oxygen - is realized by employing highly active chemical compounds . The regeneration process is controlled automatically. Upon a reduction in the amount of oxygen and increase in carbon dioxide concentration a special sensing element emits a signal, upon which an auxiliary apparatus changes the working order of the regenerator. At an excessive generation of oxygen the auxiliary mechanism is brought automatical ly into action, thus bringing about a reduction in oxygen delivery into the atmosphere of the cabin. Humidity of air is controlled in the same manner.

In case the air becomes contaminated with harmful admixtures (impurities), forming as result of live acticity of the human organism and operation of the equipment, special filters for the purification of same are provided.

Maintaining the given temperature in the ship during flight is realized by a temperature control system. It distinguishing feature is the use of a liquid coolant for the transfer of heat from the pilot's cabin, the temperature of the coolant is held at constant. The cooling agent goes from the temperature control system into a liquid-air radiator. Delivery of air through the radiator is automatically controlled depending upon the temperature of the tapping apparatus. In this way the given temperature condition in the cabin is maintained with greater accuracy.

To maintain stable temperature of the coolant and provide required temperature in the instrument section, on its outer surface is placed a radiation heat exchanger with louver system, the control of which is also automatic.

Prior to descending into a given region, the ship-satellite, before connecting the braking power plant, must acquire perfectly definite orientation in space. This problem is solved by the orientation system. In the given flight orientation is

realized by orienting one the ship's axis in direction of the Sun. The sensitive elements of that system are a number of optical and gyroscopic sensing elements. The signals entering these elements are converted in an electron block into commands regulating a system of control organs. The orientation system provides automatic scanning of the Sun, proper turn of the ship and keeping same in required position with greater accuracy.

After the ship is oriented, at a specific moment of time is cut-in the decelerating power plant. Commands for connecting the orientation system, decelerating power plant and other systems are emitted by an electron programming device.

To measure orbital parameters of ship-satellite and control the operation of its airborne equipment it is provided with a radio measuring and radiotelemetering apparatus. Measuring the movement parameters of ship and reception of telemetric information during its flight is done by ground stations, situated over the territory of the USSR . Measurement data are automatically transmitted over communication lines to computation centers , where they are processed on electronic computers. As result during the flight are obtained rough data on the basic parameter of the orbit and further movement of the ship is predicted.

The ship also carries the radio system SIGNAL, functioning on a frequency of 19.995 mc. This system serves for finding the direction of the ship and transmission of segments of telemetering information.

The TV system transmits to Earth pictures of the cosmonaut, which allows to keep visual control over his condition. One of the TV cameras transmits frontal images of the pilot, and the other one - profile.

Two way communication between cosmonaut and Earth is furnished by a radiotelephone system operating in the shortwave frequency range (9.019 and 20.006 mc) and ultrashort waves (143.625 mc).

The ultrashortwave channel is used for communication with ground points at distances of up to 1500-2000 km. Communication over shortwave channel with ground points, situated over the territory of the USSR, as shown by experience, can be secured over a larger part of the orbit.

The radio telephone system contains a magnetophone , allowing to record the conversation of the cosmonaut in flight with subsequent reproduction and transmission of same during the flight of the ship over ground receiving points. Provisions are also made for radio telegraph transmission by the cosmonaut.

The instrument panel and pilot's desk installed in the cabin are intended for controlling the operation of basic ship systems and to secure, in case of necessity, descent of ship by using manual control. On the instrument panel is situated a number of dial instruments and signal arrangement, electric clock, as well as globe , the rotation of which is synchronized with the orbital movement of the ship. The globe gives the cosmonaut the possibility to determine the current position of ship . On the pilot's desk are situated handles and on-off switches, for controlling the operation of the radio telephone system, controlling the temperature in the cabin, and for connecting manual control and decelerating power plant.

Special attention in the construction of the cosmic ship was devoted to guaranteeing flight safety.The launchings of first Soviet ships-satellites confirmed the high reliability of the performance of their equipment and devices . But on the VOSTOK ship additional steps were taken to eliminate the possibility on any accidents and to guarantee flight safety on it for man. Such a development tendency fully meets the basic problem - of creating devices allowing man to penetrate confidently into cosmic space.

To orient the ship in case of manual control the cosmonaut uses an optical orienting point, allowing to determine the position of the ship relative to the Earth.

The optical orienting device was mounted on one of pilot cabin illuminators. It consists of two annular mirror-reflectors, light filter and glass with screen. The rays coming from the line of the horizon, fall on the first reflector and then through the illuminator glass pass on to the second reflector which directs them through the glass with screen into the eye of the cosmonaut. When the ship is properly oriented with respect to the vertical the cosmonaut sees in the field of vision an image of the horizon in form of a circle.

Through the central part of the illuminator the cosmonaut watches the section of terrestrial surface directly under him. The position of the ship's longitudinal axis relative to the direction of flight is determined by observing the "course" of the terrestrial surface in the field of vision of the orienter.

Operating the control organs, the cosmonaut may turn the ship in such a way, that the line of the horizon would be visible in the orienting device in form of a concentrical ring, and the heading of the terrestrial surface would coincide with the heading line of the grid. This will indicate proper orientation of the ship. When necessary the field of vision of the orienter can be covered with light filter or blind.

The globe mounted on the instrument panel offers the possibility to determine in addition to the current position of the ship also to predetermine the point of its descent during connection of the decelerating power plant at the given moment of time.

Finally, the construction of the ship allows to descent to Earth even in the case the decelerating power plant fails. This is done by eigen (natural) deceleration of same in the atmosphere.

Supplies of food and water, regeneration substances and the volume of electric power sources are intended for a flight period of 10 days.

When constructing the ship steps were taken to prevent a temperature rise in the cabin to above specified limit when its surface is exposed to longer heating periods, which takes place at the time of gradually decelerating the ship in the atmosphere.

Medical-biological problems of man's flight into cosmic space

To solve the problem concerning the possibility of man flying in cosmic space and medical protection of same, it appeared to be absolutely necessary to:

1. to investigate the effect of cosmic flight factors on the organism, and to investigate the possible forms and method of protecting against the unfavorable effects of these factors.

2. to develop most effective methods of securing normal living conditions for the person in the cabin of the cosmic ship.

3. to develop methods for medical selection and training of cosmic ship crew members, and to develop a system of constant medical control over the state of health and life activities of cosmonauts during all stages of the flight.

Each one of the enumerated problems included a greater number of individual problems over the study and solution of which specialists in the field of physiology, hygiene, psychology , biology, clinical and professional medicine worked persistently for the past ten years. Investigations have been conducted under ground laboratory conditions and during flights of animals on rockets. Great experience, accumulated in applied fields of physiology and medicine, especially in aviation medicine and in medical protection in underwater swimming (frogmen), has been utilized. Wherever possible, special ground stands have been constructed, which allowed under laboratory conditions which allowed to investigate the effect on the organism of factors, acting in cosmic flight. The effect of overloads and organismal endurance to same was investigated on centrifugal machines - centrifuges. They reproduced accelerations ,analogous to the ones which originate during the starting of ships or during their return to

Earth.

Using vibro stands, temperature, vacuum chambers and other such installations was investigated the effect of other facters on the organism . However laboratory experiments, as a rule, could offer an answer only with respect to the effect on the organism of any one of the facters, while in real flight they affect the rocket in combination and simultaneously. Furthermore, under lab.conditions it was not possible to investigate the behavior of living organisms under weightlessness conditions. Consequently a substantial step forward toward studying the effect of cosmic flight conditions on the organism was the chance of conducting biological investigations on rockets, which began in 1951.

Several tens of experiments have been made during the flight of animals on rockets to altitudes of up to 450 km. As result of these investigations was obtained huge scientific material, characterizing reactions of physiological systems and the behavior of animals (dogs, rabbits, rats and mice) along various flight sections. Thorough examination of experimental animals during flight and for a longer period of time after their return to Earth enabled to draw a conclusion, that conditions of flying on rockets into the upper layer of the atmosphere are endured by living organisms with perfect and fullest satisfaction. Changes, noticed in individual physiological functions during the time of flight were of no painful nature,often they disappeared already in the process of experimentation leaving no aftereffects.

However, in view of the shortness of rocket flight, it was impossible to investigate the biological effect of such important cosmic flight facters as, long lasting weightlessness and cosmic radiation. Therefore, the possibility offered in 1957 to use man-made Earth satellites for biological investigations, appears to be an important step forward.

The first such experiment was carried out on the second Soviet man-made Earth

satellite. It not only confirmed but broadened the data of previous biological experiments on rockets. It was possible to prove for the first time that a longer lasting state of weightlessness in itself does not disrupt the basic vital processes.

Biological experiments were continued on the first Soviet ship-satellites. In the program of these medical-biological investigations a number of new problems was included. It appeared to be important, besides additional and more thorough investigation of the effect on the organism of long lasting weightlessness, transient conditions from weightlessness to overloads and vice versa, to conduct more thorough investigations on the biological effect of cosmic radiation. An important chapter in the program was also the study of operational characteristics and effectivenesses of systems, which in future flight were to secure normal living conditions for humans and guarantee their safe return to Earth. To bring this program into realization the first Soviet ship-satellites carried various representatives of organic world, beginning with simple forms of live to higher vertebrates.

The use in experiments of various kinds of animals and plants allowed for a full and detailed investigation of the effect of cosmic flight conditions on the most variegated processes and functions of organisms. Very broad is the information on the behavior and state of physiological functions of experimental dogs during the time of flight. The behavior of animals was observed with the aid of special TV-systems. Analysis of obtained data showed, that animals not only preserve its viability under conditions of long weightlessness and subsequent effect of overloads, but no ill effects are revealed in the state of their basic physiological functions. Thorough examination of animals after flight also revealed no deviations from normal.

Very serious attention was devoted to the detection of possible effects of cosmic radiation in flight on the ship-satellite. Numerous methods used in solving this problem revealed no changes, which could be attributed to ionizing radiation.

Results of medico-biological investigations on cosmic ships-satellites allowed to make a highly important and responsible conclusion. It was admitted, that flights on ships-satellites in an orbit, situated knowingly below the near Earth radiation belts, appear to be safe for highly organized representatives of the animal world. Results of biological experiments were used for solving the problem concerning man's flight endurance conditions.

On these bases, as well as on the bases of laboratory test results, was drawn a conclusion with respect to man's flying without ill effects to his health.

Training of cosmonauts!

The first cosmic flight could be carried out only by a person, who, realizing the enormous responsibility of the undertaking confronting him, would willingly and freely volunteer to devote all his knowledge and efforts, and possibly also his life, in the realization of this outstanding task. Thousands of Soviet citizens - patriots of their country, of various ages and professions, have expressed their willingness of flying into cosmic space. The Soviet scientists were confronted with the problem of scientifically choosing the first cosmonauts from among the large number of desiring persons.

In carrying out a cosmic flight the person is exposed to a whole complex of outer medium facters ( acceleration, weightlessness etc.), to considerable nervous emotional strains, requiring of the person mobilization of all its moral and physical forces. The cosmonaut must be able to maintain a high degree of operational activity, he must have a knowledge of orienting himself in a difficult flight condition and when necessary take over control of the cosmic ship. All this set high requirements concerning the state of health of the cosmonaut, his psychic qualities, level of his general and technical training.

These qualities are most fully combined in the profession of a pilot.

The activity of a pilot determines already the stability of the nervous-emotional sphere of the person, its good volitional qualities, and this is of special importance for first cosmic flights. Next the category of persons, participating in such flights, should be unconditionally and can be broadened considerably.

When setting up a group of cosmonauts interviews were held with a greater number of fliers expressing the desire to carry out the cosmic flight. The best ones prepared of these underwent thorough clinical and psychological examinations. The purpose of such an examination was: to determine the condition of health, to reveal hidden deficiencies or reduced organismal stability to individual factors, characteristic of the forthcoming flight, to evaluate the reaction of the person during the effect of these factors.

Examinations were made with the use of numerous modern biochemical, physiological electrophysiological and psychological methods and special functional tests, allowing to estimate the reserve possibilities of basic physiological systems of the organism (investigation in the barometric chamber at considerable degrees of air rarefaction, at barometric pressure drops and breathing oxygen at higher pressure, investigation on the centrifuge etc.).

An important phase was psychological examination, which was directed for the exposure of persons, having finest memory, alertness, active easily changing attention, capable of rapidly developing accurate coordination movements.

As result of the clinical-physiological examinations was formed a group, which began carrying out the program of special training, training on special stands and trainers, imitating under ground and flight conditions the factors of cosmic flight. Simultaneously were determined the individual reactio characteristics of the organism to the effect of imitated factors.

The special training program was intended so that the cosmonauts would acquire

necessary data on basic theoretical problems connected with problems of the forth coming flight, as well as practical habits in the use of the equipment and devices in the cabin of the cosmic ship. This program called for studying the bases of rocket and cosmic technology, construction of cosmic ship, special astronomical problems, problems of geophysics, foundations of cosmic medicine.

The program of special training and testing included:

Fig.169.Yu.A.Gagarin before flight

flights on aircraft under weightlessness conditions; training in a model cabin of a cosmic ship and on a special trainer; longer stay in a specially equipped sound proof chamber; training on a centrifuge; parachute jums from aircraft

During the execution of special training tasks were also solved certain problems of securing the condition for man's cosmic flight, particularly such problems



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connected with feeding the cosmonaut in flight, his clothing, with air regeneration system.

During flights on aircraft were investigated the individual reactions of cosmonauts during the effects of weightlessness and transition from weightlessness to overloads. Investigated was the possibility of carrying on radio communication, receiving water and food and so on. This made it possible to reply to numerous important questions on the possible actions of man under cosmic flight conditions.

It was established, that all selected cosmonauts fare well in the state of weightlessness. Furthermore, it was established, that under weightlessness conditions lasting 40 seconds, it is possible to receive liquid, semi-liquid and solid food, execution of small coordinated acts ( writing, purposefull movements of hand), carrying on radio communication, reading and visual orientation in space as well).

Training in a make believe cosmic ship cabin and on special trainer were carried out for the purpose of studying the equipment and evices of the cabin, development of variants of flight mission, adaptation to being in a real cabin of a cosmic ship. For this was created a special stand-trainer which with the aid of electron-modulating devices allowed to reproduce on the instruments the real changes, corresponding to such in flight. The actions of the pilot also corresponded to real ones. Provisions were made for imitating unusual (emergency) flight variants and to train the cosmonaut at similar situations.

The main problem of investigations during longer stay in a specially equipped sound proof chamber was to determine the nervous-psychic stability of the cosmonaut during his longer stay in isolated space of limited volume, all by himself, at a considerable reduction of outer stimuli. Conditions of day and feeding conditions have been reproduced, close to the ones, which will take place in real flight.

A larger scope of physiological investigations, and special psycho-physiological methods enabled to reveal persons with higher precision qualities, accuracy in executing missions, possessing a more stable nervous-emotional sphere.

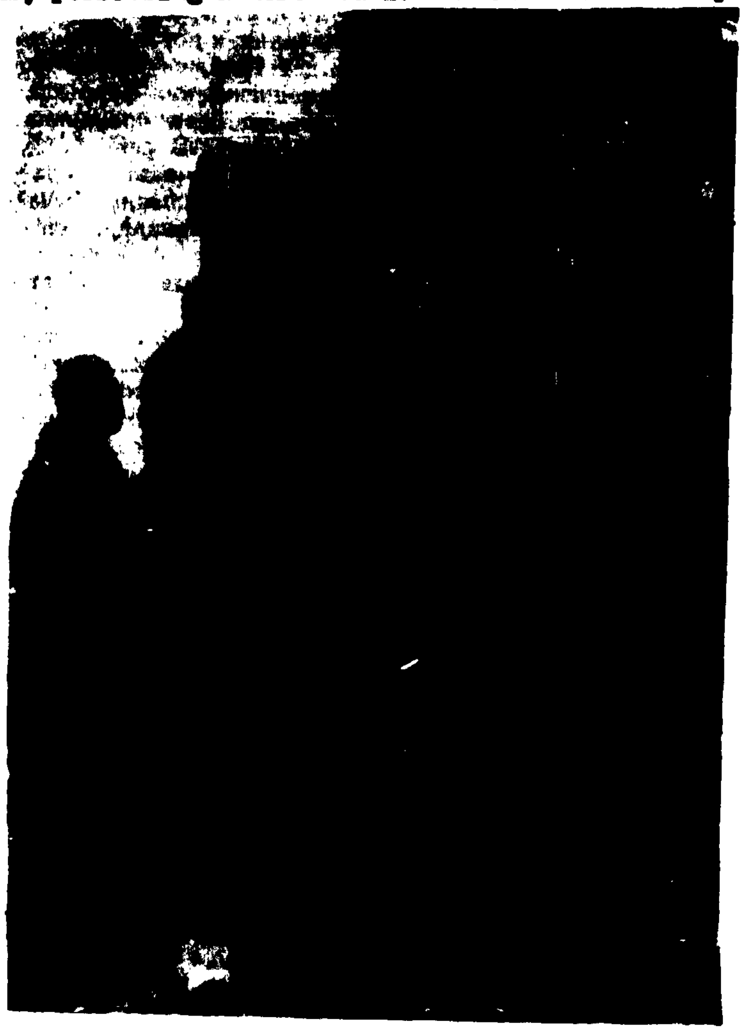

Fig.170. On the starting platform before the flight of ship VOSTOK into cosmos.

When training on the centrifuges, in thermal chambers, was determined the individual endurance of the cosmonaut to corresponding effects, their effect on the process of basic physiological functions was investigated, problems of increasing the resistance of the organism to factors created by outer medium have been solved. It was established on the basis of results, that cosmonauts have excellent stability



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against the effects of above mentioned factors, persons were revealed, which withstood the tests better than others.

In the parachute training process each cosmonaut executed several tens of jumps.

Physical training of the cosmonaut group consisted of planned occupations and early morning exercises. Planned occupations were carried out with consideration of individual characteristics of physical development of each cosmonaut. Early morning exercises were conducted daily for one hour and its purpose was general physical training. Physicocultural exercise were intended for the purpose of raising the stamina of the organism to the effects of accelerations, development and improvement of habits of freely mastering the body in space, raising the ability of enduring long lasting physical stresses.

Physical training was executed under constant surveillance of a doctor (MD) who selected especially fitting gym exercises, games, jumping into the water, swimming and physical exercises on special installations.

After completing the special training program was organized direct training for the forthcoming cosmic flight. This training included :

study of flight missions, study of maps of landing region, piloting instructions, carrying on Radio communication etc ;

studying emergency supply, utilization of same at point after landing, study of direction finding system etc;

centrifuge testing in protective clothing at maximum values of anticipated overloads;

long lasting testing in a make believe cosmic ship using all the systems provising the conditions for active life.

As result of the scientific-training operations was selected a group of cosmonauts, ready for flight into cosmic space.

To bring man's first cosmic flight into realization from the group of cosmonauts was selected pilot major Yuriy Alekseyevich Gagarin.

The remarkable Soviet citizen Yu.A.Gagarin, was born on March 9, 1934 in the hosehold of a collective farmworker . His long dream was to become a flier. Having finished in 1957 the Orenburg Aviation School, he graduated as military fighter pilot specialist. Yu.A.Gagarin served in one of the Armed Forces Components of the USSR Upon his insisten appeal he was included in the group of cosmonaut candidates and successfully passed the selection. When the cosmonaut group was in training Yu.A. Gagarin was one of the best.

The great confidence in being the first in the world pilot-cosmonaut was fully justified by the capabilities of Yuriy Alekseyevich Gagarin.

First Cosmic Flight

The blast off of the cosmic ship VOSTOK took place on April 12, 1961 at 9 hrs 7 min. Moscow time.

During the entire time of orbiting pilot-cosmonaut Yu.A.Gagarin maintained continuous radio telephone communication with the ground flight vectoring center. He transmitted precise data on the changes in overloads at the moments the stages separated themselves from the carrier rocket. The noise in the cabin of the ship was not greater than in a cockpit of a jet aircraft. Already at the time of orbiting Yu.A.Gagarin observed the Earth in the illuminators .

Operation of board equipment during flight in orbit, orientation and descent of ship were controlled automatically. However, in case of necessity, the cosmonaut upon his own desire or upon command from Earth could take ship control into his own hands, determine its position and descent into selected region.

Weightlessness came immediately after the ship came into orbit. At first this condition felt like something unusual for the cosmonaut, but soon he became used

to it and did not mind it.

In conformity with the mission and flight program he watched the operation of ships equipment, he was in constant telephone and telegraph radio communication with the Earth, did his observations in the illuminators and in the optical orienter, reported to Earth and dotted down observation data into the log book on board the ship and on a magnetic tape, took food and water.

The pressure in the cabin of the ship during flight was maintained at 750-770 mm Hg, air temperature at 19-22°C, relative humidity 62-71%.

The surface of the Earth was well surveyed from an altitude of up to 300 km. Very well visible were the shore lines, larger rivers, contour of the Earth's surface, forests, clouds and shadows from the clouds. When flying over the territory of our country Yu.A.Gagarin observes the larger stretches of collective farms.

The sky - absolutely black. The stars on it looked much brighter and were more clearly visible, than from the Earth. The Earth has a very beautiful bluish halo. Colors on the horizon change from soft blue, through blue, gray, violet - to black color of the sky. When coming out from the shadow at the horizon of the Earth one could observe a bright orange color, which then changed into all the colors of a rainbow.

At 9 hrs, 51 min, the automatic orientation system of the ship was cut in. After coming out from the shadow scanning function was carried out and the ship was oriented toward the Sun.

At 9 hrs, 52 min cosmonaut Yu.A.Gagagrin flew over the region of Cape Horn, reported about his good state of health and normal functioning of the equipment on board the ship.

At 10 hrs, 15 min from the automatic programming installation went out commands for readying the board equipment for connection of the decelerating power plant.

At that moment the ship approached Africa and from Yu.A.Gagarin arrived a message about the process of the flight.

At 10 hrs. 25 min. the decelerating power plant was connected and the ship changed into orbit of an Earth satellite following a descending trajectory.

At 10 hrs, 35 min the ship began entering the dense layers of the atmosphere.

Having completed the first in thr world cosmic flight with a cosmonaut on board, the ship-satellite VOSTOK landed at a given region at 10 hrs. 55 min.Moscow time.

Fig.171. Yy.A.Gagarin reports over the telephone to comrade N.S.Khrushchev about his safe landing.



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The well being of Yu.A.Gagarin during all the stages of flight was satisfactory He maintained complete freedom of operation.

Highly interesting are the data, characterizing the change in the condition of the cardio-vascular and respiratory systems of the cosmonaut. During the pre-blast off period Yu.A.Gagarin's pulse frequency was 66 beats per minute, and the respiration frequency - 24. At the orbiting section the pulse frequency rose to 140-158, and respiration frequency was 20-26. When weightlessness came the cardio-vascular and respiratory system indices gradually came down to the initial values. At the tenth minute of weightlessness pulse frequency was 97, respiration - 22.

During the descent under the effect of overloads short periods of respiratory increases have been noticed. However when approaching Earth respiration became calm with a frequency of about 16 per min. Three hours after the landing the pulse frequency was 68, respiration - 20 per min, which corresponds with the normal condition of Yu.A.Gagarin.

After returning from cosmic flight Yu.A.Gagarin feels fine . There are no disorders in the state of his health.

The first in the history of humanity flight into cosmic space, carried out by Soviet cosmonaut Yu.A.Gagarin on board the ship-satellite VOSTOK, allowed to make a conclusion of enormous scientific importance about the practical possibility of man's flight into cosmos. He has shown, that a person can normally endure conditions of cosmic flight, conditions of orbiting and return to the surface of the Earth. This flight has proven that under conditions of weightlessness a person can fully retain freedom of operation, motional coordination, clearness of thought.

The flight furnished extremely valuable data on the performance of the construction and equipment of the cosmic ship in flight. Full confirmation has been furnished of the scientific and technical solutions, on which that construction

was based. Approval was given to the reliability of the carrier-rocket and structural perfection of the ship-satellite.

## Conclusion

The creation of man-made satellites, launchings of cosmic rockets toward the Moon and Venus, flights of first cosmic ships with return to Earth, realization of the first in the history of cosmic flight man's flight have heralded the advent of an era when interplanetary space will be mastered.

Only three years have passed from the time when the first man-made celestial body was made - the first Soviet man-made Earth satellite came into being. During this relatively short period of time we have witnessed a whole series of grandiose scientific and technical achievements, which only recently appeared to be a far off dream.

In our time science and technology are making annually greater and greater strides. The technology of cosmic flights is developing extremely intensively.

The flight of Soviet cosmic ship VOSTOK carrying on board the first cosmonaut major Gagarin Yu.A. appears to be an unprecedented victory of man over the forces of nature, a decisive step in man's penetration into the cosmos.

Cosmic devices of various types will find broader and broader application in solving various scientific and practical problems.

The time has come for practical realization of the hitherto fantastic projects - the time for the creation of extraterrestrial scientific stations, cosmic journeys of man to the Moon, Mars, Venus and other planets of the solar system, and then even beyond their limits.

Solution of these grandiose problems requires enormous collective creational forces, further development of many fields of theoretic and experimental science, many branches of technology.

The Soviet Union has firmly taken the lead in the realization of cosmic flights.

Rapid development of science and technology in our country and the creative enthusiasm of our people, guided by the Communist Party, are a reliable guarantee, that in the field of man conquering the cosmos priority and leading position henceforth belong to the USSR.

## Appendices

### Appendix 1.

Basic dependencies of motion in the central field of gravitation.

Integration of equations of motion of the material point in central field of gravitation

Equations of motion have the form of (see chapter 1):

$$\frac{1}{2}(\dot r^2 + r^2\dot\varphi^2) - \frac{K}{r} = \frac{1}{2}V_0^2 - \frac{K}{r_0} \qquad (1)$$

$$r^2\dot\varphi = V_0 r_0 \cos\vartheta_0. \qquad (2)$$

According to the second equation

$$\dot\varphi = \frac{V_0 r_0 \cos\vartheta_0}{r^2}, \qquad (3)$$

but $\dot\varphi$ can be presented in form of

$$\dot\varphi = \frac{d\varphi}{dt} = \frac{d\varphi}{dr}\cdot\frac{dr}{dt} = \frac{d\varphi}{dr}\cdot\dot r \qquad (3a)$$

and consequently

$$\dot r = \frac{dr}{d\varphi}\,\frac{V_0 r_0 \cos\vartheta_0}{r^2}. \qquad (4)$$

Having substituted (3) and (4) in equation (1), we will obtain:

$$\left(\frac{dr}{d\varphi}\right)^2 \frac{V_0^2 r_0^2 \cos^2\vartheta_0}{r^4} + \frac{V_0^2 r_0^2 \cos^2\vartheta_0}{r^2} - \frac{2K}{r} = V_0^2 - \frac{2K}{r_0}, \qquad (4a)$$

hence

$$\frac{dr}{d\varphi}\cdot\frac{V_0 r_0 \cos\vartheta_0}{r^2} = \sqrt{V_0^2 - \frac{2K}{r_0} - \frac{V_0^2 r_0^2 \cos^2\vartheta_0}{r^2} + \frac{2K}{r}}. \qquad (4b)$$

or

$$d\varphi = \frac{-d\left(\frac{V_0 r_0 \cos\vartheta_0}{r}\right)}{\sqrt{V_0^2 - \frac{2K}{r_0} - \frac{V_0^2 r_0^2 \cos^2\vartheta_0}{r^2} + \frac{2K}{r}}} \quad (4c)$$

For the convenience of integration we will introduce a constant value $\frac{K}{V_0 r_0 \cos\vartheta_0}$ under the sign of the differential in the numerator, and we will also add and compute the value $\frac{K^2}{V_0^2 r_0^2 \cos\vartheta_0}$ in the radicant.

As result we will obtain:

$$d\varphi = \frac{-d\left(\frac{V_0 r_0 \cos\vartheta_0}{r} - \frac{K}{V_0 r_0 \cos\vartheta_0}\right)}{\sqrt{\left(V_0^2 - \frac{2K}{r_0} + \frac{K^2}{V_0^2 r_0^2 \cos^2\vartheta_0}\right) - \left(\frac{V_0 r_0 \cos\vartheta_0}{r} - \frac{K}{V_0 r_0 \cos\vartheta_0}\right)^2}} \quad (5)$$

Integration of this formula leads to equation of the orbit:

$$\varphi - \varphi_0 = \arccos \frac{\frac{V_0 r_0 \cos\vartheta_0}{r} - \frac{K}{V_0 r_0 \cos\vartheta_0}}{\sqrt{V_0 - \frac{2K}{r_0} + \frac{K^2}{V_0^2 r_0^2 \cos\vartheta_0}}} \quad (6)$$

where $\varphi_0$ - integration constant - depends upon the beginning of calculating angles $\varphi$.

Equation of orbit (6) can be presented in form of :

$$r = \frac{\frac{V_0^2 r_0^2 \cos^2\vartheta_0}{K}}{1 + \left(\frac{V_0 r_0 \cos\vartheta_0}{K}\right)\sqrt{V_0^2 - \frac{2K}{r_0} + \frac{K^2}{V_0^2 r_0^2 \cos\vartheta_0}}\cos(\varphi - \varphi_0)} \quad (7)$$

or introducing designations

$$p = \frac{V_0^2 r_0^2 \cos^2\vartheta_0}{K} \quad (8)$$

$$e = \frac{V_0 r_0 \cos\vartheta_0}{K}\sqrt{V_0^2 - \frac{2K}{r_0} + \frac{K^2}{V_0^2 r_0^2 \cos^2\vartheta_0}}, \quad (9)$$

in form of :

$$r = \frac{p}{1 + e\cos(\varphi - \varphi_0)}. \quad (10)$$

## Determining Period of Rotation over Elliptical Orbit

The sectorial velocity of the material point, moving over an elliptical orbit in the central field of gravitation, is constant and equals $\frac{1}{2} V_0 r_0 \cos\vartheta_0$ (see chapter 1), and the area encompassed by the elliptical orbit (area of the ellipse), is $\pi ab$, where a and b - larger and smaller semiaxis of the ellipse respectively.

Consequently, the period of rotation (round trip) (time of full rotation of the point in orbit) will be equal:

$$T = \frac{2\pi ab}{V_0 r_0 \cos\Theta_0}. \qquad (11)$$

For the ellipse we have

$$b = a\sqrt{1-e^2} \qquad (11a)$$

and

$$a = \frac{p}{1-e^2}. \qquad (11b)$$

hence

$$1 - e^2 = \frac{p}{a}. \qquad (11c)$$

and consequently

$$b = a\sqrt{\frac{p}{a}}. \qquad (12)$$

Having substituted in (11) the value b from (12) and p from (8) we will obtain after elementary transforms a formula, determining the period of rotation:

$$T = 2\pi\sqrt{\frac{a^3}{K}}. \qquad (13)$$

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